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RESEARCH MEMORANDUM

REVIEW OF STATUS, METHODS, AND POTENTIALS OF

GAS-TURBINE AIR-COOLING

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON

February 23, 1955

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SUMMARY

The use of turbine cooling to allow increased turbine-inlet temperatures in gas-turbine engines permits greatly increased engine power output (thrust or shaft horsepower) and, for some types of engines, permits improved specific fuel consumption. In addition, cooling allows a greater degree of freedom in turbine design because of higher permissible stress levels and a greater range of possible turbine materials. The attainment of these benefits from turbine cooling is accompanied by a small performance reduction relative to the ideal uncooled engine performance as a result of cooling losses. For nonafterburning turbojet engines operating at a flight Mach number of 2 and constant turbine-inlet temperature, approximately a 1-percent reduction in thrust accompanies each percent of air bled from the compressor exit for turbine cooling. The effect on specific fuel consumption is generally negligible. The prevalent practice of bleeding air overboard for cabin cooling, driving accessories, and so forth, results in a thrust reduction more than double that caused by turbine cooling and causes substantial increases in specific fuel consumption. The power reduction resulting from air-cooling a turboprop engine at subsonic speeds and constant turbine-inlet temperature is somewhat higher than for a turbojet engine, but the net gains in power resulting from higher turbine-inlet temperatures are still very large. In addition, operation at the higher turbine-inlet temperatures can result in an actual decrease in specific fuel consumption, including the effects of cooling, relative to low-temperature uncooled engines.

At present, the most promising air-cooled turbine blades for conventional turbojet engines are convection-cooled corrugated-insert and strut-supported blades. Successful analytical methods of predicting blade temperatures have been found for both blades. General heat-transfer analyses indicate that these blades will probably be satisfactory except for operation under the combination of very high gas temperatures (2500° F) and high flight Mach numbers (above 2), where transpiration-cooling may be required. For small or thin blades, the maximum permissible temperatures and flight Mach numbers may be lower.

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INTRODUCTION

Past history has shown that, in order to obtain desirable cycle temperatures for heat engines, it was often necessary to cool certain engine components to circumvent material strength limitations. Thermodynamically, removal of heat from a cycle by cooling is detrimental to performance; but, practically, cooling permits a type of engine operation resulting in performance unattainable without cooling. An excellent example is the piston engine, where refinements in cooling methods, involving the removal of large quantities of heat, led to increasingly superior performance. It can be expected that a similar type of evolution will occur in the cooling of gas-turbine engines and result in a type of operation presenting many new performance possibilities.

There are three main purposes for cooling the turbines of gas-turbine engines. The first and most commonly accepted purpose is that higher engine cycle temperatures (resulting in increased specific power) can be obtained if means are provided for controlling the turbine disk and blade temperatures independently of the turbine-inlet gas temperature. The second purpose is to permit the use of higher operating stresses or to give longer turbine life by reducing the operating metal temperature of the turbine. Higher operating stresses allow a much greater amount of freedom in the turbine design and generally result in increased power for a given engine frontal area, because the flow capacity of the turbine can be increased through the use of longer turbine blades. The third purpose of turbine cooling is to permit a greater degree of freedom in the choice of turbine materials than is presently available to engine designers. Materials currently used for gas turbines are chosen for strength characteristics at high operating temperatures. In general, these materials also contain relatively large quantities of scarce or critical alloying elements. Lowering the turbine blade temperature permits the use of other materials that, in addition to having the capacity of withstanding higher stress levels, have reduced amounts of critical alloying elements relative to the so-called "high-temperature alloys."

This report discusses the use of less critical turbine materials and presents some of the performance possibilities attainable when turbine cooling is employed to permit use of higher cycle temperatures and higher turbine stresses. In addition, some of the progress that has been made in turbine air-cooling research is described, and an indication is given as to the expected trends in the development of cooled gas-turbine engines.

POTENTIAL ENGINE PERFORMANCE AT HIGHER TURBINE-INLET TEMPERATURES

Turbojet Engines

The thrust of a turbojet engine is essentially proportional to the product of the weight flow of air that passes through the engine and the velocity of the jet at the exhaust nozzle. All methods of increasing the thrust output, therefore, depend upon increasing either one or both of these variables in some manner. The weight flow of air per unit frontal area can be increased through the use of recently developed compressors. The jet velocity can be increased through the use of (1) higher compressor pressure ratios, higher turbine efficiencies, or lower burner pressure losses, all of which result in a higher available pressure in the exhaust nozzle and consequently allow higher expansion ratios for obtaining higher velocities, (2) improved exhaust-nozzle efficiencies, and (3) increased jet-exhaust temperatures.

Several of these effects on engine performance are illustrated in figure 1 (data from ref. 1) for both nonafterburning and afterburning turbojet engines operating at a flight Mach number of 2 in the stratosphere. With the newer compressors, the flow per unit of engine frontal area is often determined by the turbine. For this reason, the relative engine thrust is given per unit of turbine frontal area in figure 1. For this case, high compressor pressure ratios result in higher gas densities at the turbine and consequently increased flow capacity. (The compressor pressure ratios shown are for the actual operating conditions and not sea-level static.)

It can be seen from the figure that, for nonafterburning engines, turbine-inlet temperature has a very significant effect on thrust. Increases in turbine-inlet temperature can increase the thrust by a factor of 2 or more relative to that of present engines. At constant compressor pressure ratio, increases in turbine-inlet temperature generally result in increased specific fuel consumption. These increases can be explained by the fact that the thrust is increased directly as the jet velocity is increased, but the kinetic energy of the gases is increased as the square of the jet velocity. The fuel consumption is proportional to the kinetic-energy increase; therefore, the fuel consumption increases at a greater rate than the thrust, with a resultant increase in thrust specific fuel consumption. This effect can be largely compensated for by increasing the engine thermal efficiency through the use of higher compressor pressure ratios.

Figure 1 also shows that very high thrust levels are obtainable for afterburning engines. This thrust is obtained at a relatively high cost in specific fuel consumption, particularly at the turbine-inlet temperatures of current engines. If the turbine-inlet temperature is increased, a smaller pressure drop is incurred across the turbine, less fuel is burned in the exhaust nozzle (although more is burned ahead of the turbine), and the thrust output of the engine is increased at a substantial saving in specific fuel consumption. The advantage of increased turbine-inlet temperatures for afterburning engines is therefore primarily to decrease the fuel consumption. In addition, higher compressor pressure ratios combined with high turbine-inlet temperatures give very large increases in thrust and improved specific fuel consumption.

Turboprop Engines 3511

The power output of the turboprop engine, or other shaft-power turbine engines, is primarily a function of energy level of the gases ahead of the turbine; therefore, the turbine-inlet temperature has a direct bearing on the power. In the turboprop engine the gases are expanded almost completely in order to extract the maximum power, so that the pressure level and density at the last stage of the turbine are always low. Therefore, turbine flow capacity (weight flow per unit turbine frontal area) is almost independent of compressor pressure ratio. Because of this fact, power per unit turbine frontal area has little significance; of greatest interest is power per pound of compressor air, defined as specific horsepower.

The effects of turbine-inlet temperature and compressor pressure ratio on relative specific horsepower and specific fuel consumption are shown in figure 2. Here, as in figure 1 for the turbojet engine, the power output can be increased by a factor of 2 or more relative to present engines by increasing the turbine-inlet temperature. It will be noted, however, that opposite to the case for the nonafterburning turbojet, increasing turbine-inlet temperature decreases specific fuel consumption for the turboprop. A rigorous explanation of this trend is somewhat involved, but basically the reason for the decrease in specific fuel consumption with increasing turbine-inlet temperature is that both the gross turbine power and the fuel-flow rate are directly proportional to the turbine-inlet temperature. The net turbine power, or shaft power, is the gross turbine power minus the compressor power, and therefore it increases with turbine-inlet temperature at a rate proportionally greater than for the gross power. As a result, the specific fuel consumption, which is the ratio of fuel-flow rate to shaft power, decreases with increasing turbine-inlet temperature.

DESIGN FREEDOM OBTAINABLE WITH TURBINE COOLING

The flow capacity of modern compressors is rapidly increasing to the point where components other than the compressor (inlet diffuser, primary burner, turbine, afterburner, or exhaust nozzle) will determine the engine frontal area. For some applications, it appears that the turbine may have the largest diameter of any component of the engine. The flow area of the turbine can be increased by the use of longer turbine blades, but this often results in stresses in excess of those permissible with presently available materials in uncooled turbines. An indication of the manner in which turbine diameter is related to turbine blade stress for a given compressor weight flow and pressure ratio is shown in figure 3. Increasing the blade root stress from 30,000 to 60,000 pounds per square inch can result in a reduction in single-stage turbine diameter of approximately 15 percent. This corresponds to a frontal-area reduction of over 25 percent.

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Properties of some materials that can be used to obtain higher turbine stress levels are indicated in figure 4. For the temperatures at which gas-turbine blades operate, stress-rupture is the criterion that usually determines allowable blade stress. The stress-rupture properties of several materials are shown, and the upper levels of the curves are cut off where stress-rupture properties no longer determine the permissible stress. The alloy S-816 is commonly used in present gas-turbine engines. At a temperature of 1500° F (about standard blade temperature for present engines), the maximum allowable stress is 24,000 pounds per square inch. If, however, the temperature is reduced only 100° F by cooling, the allowable stress can be increased by about 35 percent, with further increases obtainable at lower temperatures. Below temperatures of about 1200° F, however, other materials, such as A-286, possess better strength properties, with the possibility of operating at stresses over 90,000 pounds per square inch - over $3\frac{1}{2}$ times the allowable stress for present engines.

Further reduction in temperature makes possible the utilization of high-strength steels such as Timken 17-22A(S). This type of material offers only slight increases in possible operating stress over A-286, but the critical-material content of 17-22A(S) is almost completely eliminated, the alloy containing about 97 percent iron. Even A-286 is presently considered a relatively noncritical alloy, because it is over 50 percent iron and contains no cobalt or columbium. Currently used blade materials such as S-816 contain very high quantities of critical materials such as cobalt, nickel, chromium, and columbium, the iron content of S-816 being only 2.8 percent.

In general, materials that exhibit high strength at temperatures from 1000° to 1300° F contain considerably smaller quantities of critical materials than the currently used high-temperature alloys. The significance of this fact is that cooling, in addition to permitting the use of higher turbine-inlet temperatures and stresses in turbine engines, permits the production of a greater number of engines because of the greater availability of suitable turbine materials. The factor of material availability was the primary reason the Germans used turbine cooling in some of their turbojet engines in World War II.

The very high stresses apparently available through the use of turbine cooling (fig. 4) do not necessarily mean that it is possible to operate the turbine blades at these stress levels. As discussed in references 2 and 3, a factor of safety, defined as a stress-ratio factor, is required in the design of air-cooled turbine blades. This factor of safety differs little from that used in standard design practice, except that the actual stress-rupture properties of the materials in air-cooled turbine blades are not known after blade fabrication. The stress-rupture properties shown in figure 4 are for bar stock. In air-cooled turbine blades, the metal is thin and the structures are usually brazed; both the thin metal wall and the brazing have the effect of decreasing the stress-rupture properties. This effect is believed to vary greatly with materials and brazing alloys and techniques used. Reference 3 showed that, for one type of blade design made of alloy 17-22A(S), the required stress-ratio factor was 2.3. This is the ratio of the stress-rupture for bar stock to the centrifugal stress in the failure area of the air-cooled turbine blade. It is believed that the use of other blade materials and possibly of other fabrication techniques may result in much lower values of stress-ratio factor. Even with high values of stress-ratio factor, higher turbine operating stresses are permissible through the use of turbine cooling than are presently possible with uncooled turbine blades.

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EFFECT OF TURBINE COOLING ON ENGINE PERFORMANCE

The most probable source of air for turbine cooling is the engine compressor. It is often possible to bleed the compressor air from some intermediate stage in order to keep the cooling-air temperature and the compressor work on the cooling air at a minimum, but for turbine rotor cooling it is more often necessary to bleed from the discharge of the compressor in order to obtain the highest possible pressure for effective blade cooling. The use of this air for cooling affects engine performance in several ways. The air is removed from part of the engine cycle so that it is unavailable for doing work in the turbine, but the work done on the cooling air makes the required specific turbine work higher than without bleed. When the air is used for turbine rotor cooling, additional work is done on the cooling air to accelerate it to the wheel speed at the blade tip as it passes through the rotor. For a given value of turbine-inlet temperature, therefore, the pressure and temperature ratios across the turbine will be somewhat higher when air is bled from the compressor.

Reduction of the exhaust-gas temperature due to dilution by the cooling air results in a thrust reduction compared with the case with no cooling air, but the mass-flow addition in the exhaust jet due to the cooling air results in higher thrusts than if the air were not mixed into the exhaust gases.

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Turbojet-Engine Performance

Present knowledge of the quantities of cooling air required for operation at higher turbine-inlet temperatures may be used to predict the engine performance attainable with air-cooled turbines. Turbojet-engine performance with no consideration given to the effects of bleeding part of the compressor air for turbine cooling was discussed in connection with figure 1. The combined effect of increasing the turbine-inlet temperature and bleeding air from the discharge of the compressor to cool the turbine is shown in figure 5. These analytical results are based on the coolant flows that were calculated to be required for single-stage turbines with corrugated-insert blades. The total coolant flows for both turbine stator and rotor were approximately 3 percent at 2000° F, and 8 percent at 2500° F. These are probably about the minimum flows that could be expected with corrugated-insert blades, but should be easily within the capabilities of other blades to be discussed later. Cooling generally results in a small decrease in thrust with practically no sacrifice in fuel consumption relative to the calculated performance without cooling for nonafterburning engines. In afterburning engines, the effect of cooling is to increase fuel consumption with only small effects on thrust. These effects can be explained by the fact that cooling generally shifts the performance map in the direction of lower turbine-inlet temperatures by diluting the exhaust gases and lowering the temperature downstream of the turbine.

Turboprop-Engine Performance

Air-cooling. - The predicted performance of an air-cooled turboprop engine at sea-level static conditions is shown in figure 6. The cooling-air requirements were assumed to be the same as for the turbojet engine. The decrease in performance relative to the calculated performance without cooling (fig. 2) is greater for the turboprop than for the turbojet engine, because in the turboprop the cooling air passing through a cooled turbine stage is completely lost for engine power generation in that stage of the turbine. In the turbojet engine, on the other hand, the cooling air is still available for obtaining power in the form of jet thrust. At turbine-inlet temperatures up to 2000° F with the turboprop engine, however, the specific fuel consumption decreases with increasing values of turbine-inlet temperature even when the effects of cooling are included.

Liquid-cooling. - Also shown in figure 6 is a performance point for a liquid-cooled turbine under the most severe conditions given. In this case the only effect of cooling on the cycle is the removal of a small portion of heat from the gases. The effect on performance is extremely small; therefore, liquid-cooling of turboprop engines is very promising if radiator drag losses are tolerable. Whether the radiators will cause an acutal drag is open to question and will be determined by the radiator design. The airplane design will, of course, be materially affected by requirements for ducting air aboard for the radiator. For stationary power plants, the radiator usually does not present a problem; consequently, liquid-cooling is extremely promising for this use. Even though the performance of air-cooled turboprop engines is inferior to the performance of liquid-cooled engines, the use of air-cooling is very promising for aircraft, because substantial increases in power are attainable at no increase in specific fuel consumption relative to uncooled engines at current gas-temperature levels, and no radiators are required. 3511

Performance Variations with Compressor-Discharge Air Bleed

Previous figures have shown the performance potentialities attainable through the use of turbine cooling to permit engine operation at higher turbine-inlet temperatures. It is also important to study the engine performance variations resulting from turbine cooling at a constant turbine-inlet temperature for various quantities of cooling air to determine the effort that must be expended in finding methods that will require smaller quantities of coolant and that will reduce losses.

The effect of bleeding various amounts of air from the compressor for cooling or other purposes is shown in figures 7 and 8 for nonafterburning turbojet and turboprop engines, respectively. The quantity of air required for cooling depends upon the type of air-cooled blades used in the engine; therefore, at given engine conditions a wide variation in cooling-air requirements is possible. In most aircraft gas-turbine engines, air is bled from the compressor for such uses as accessory drives or cabin cooling, in which cases the air is thrown overboard and cannot be used for jet thrust or turbine power. As a basis of comparison, the effects on performance of this prevalent practice of overboard bleed are shown in addition to the effects of bleeding air for turbine-cooling purposes.

The variations of relative specific thrust and specific fuel consumption with various percentages of cooling air bled from the compressor discharge for a turbojet engine operating at a flight Mach number of 2 in the stratosphere are shown in figure 7. The same general trends are obtained at conditions other than those given for this figure, except that, at lower turbine-inlet temperatures, losses due to cooling are somewhat higher. The thrust decreases approximately 1 percent for every percent of air bled from the compressor for turbine-cooling purposes. This loss is more than doubled if the air is bled overboard and cannot be used for jet thrust. Air bled for turbine cooling has only a very slight effect on specific fuel consumption, but overboard bleed increases the specific fuel consumption over 2 percent for every percent of bleed air.

The effect of bleeding various amounts of cooling air on the performance of a turboprop or a stationary power plant operating at sea-level static conditions (no jet thrust horsepower) is shown in figure 8. As explained in connection with figure 6, the decrease in performance due to air-cooling at a constant turbine-inlet temperature is larger for a turboprop than for a turbojet engine. This is particularly true concerning specific fuel consumption, for which the increase with cooling-air flow is more than 5 times as high as for a turbojet engine. The decrease in power due to bleeding air for turbine cooling is only slightly higher than the decrease in thrust for a turbojet. As stated previously, however, operation at the higher turbine-inlet temperatures permitted by turbine cooling can result in an actual decrease in specific fuel consumption, including the effects of cooling, relative to low-temperature uncooled engines. For turboprop engines, bleeding air for turbine cooling is only slightly less costly than for cabin cooling, accessory drives, electronic equipment cooling, and so forth.

As previously discussed, the effect of liquid-cooling on turboprop or stationary power-plant performance is extremely small. The change in power or specific fuel consumption caused by liquid-cooling at the conditions of figure 6 is less than 0.5 percent. For aircraft application, of course, the effect of radiator drag on over-all performance would also have to be included.

As explained in reference 4, there is an optimum jet velocity for turboprop engines at each flight speed that results in minimum fuel consumption and maximum thrust. The performance variations due to bleeding air from the compressor for cooling purposes are somewhat smaller for turboprop engines with optimum jet thrust than for the case with no jet thrust, but the improvement is of little significance.

Up to the present time it has been impossible to verify experimentally all the effects of cooling on engine performance. Probably the largest unknown in the prediction of engine performance is the effect on turbine aerodynamic performance of discharging cooling air from the blade tips. Tests at the NACA, while not conclusive, do not show a measurable effect of the cooling-air discharge on turbine efficiency. Tests conducted by the British (ref. 5) show that a coolant flow of 2 percent of the compressor flow to each the turbine rotor and the turbine stator affected the turbine stage efficiency less than 0.5 percent. In the British investigation, the stator cooling air was discharged at the stator inner diameter ahead of the turbine rotor. If, instead, this air had been ducted to mix with the exhaust gases downstream of the turbine, an even smaller effect on turbine efficiency could be expected.

The effect of discharging the rotor cooling air through several blade tip configurations was investigated in reference 6, which indicated that with the proper tip design a substantial portion of the pumping work on the cooling air in the turbine rotor can be recovered in the form of added turbine work. Generally, the effect of this turbine work on the over-all performance is of rather small significance, as illustrated in figures 7 and 8. The difference between the stator and rotor cooling curves is due to the pumping work in the turbine rotor. The experimental studies just mentioned indicate that the methods used for predicting cooled-engine performance are reasonably accurate and the trends shown should be correct.

RESEARCH ON AIR-COOLED TURBINE BLADES

Both air- and liquid-cooling methods have their relative advantages and disadvantages. Most NACA research on liquid-cooling has been of a fundamental nature to determine the laws governing heat transfer within the cooling passages. This research is summarized in reference 7, and more recent research on natural-convection cooling is presented in reference 8. Liquid-cooling research on actual turbine blade configurations and systems has been very limited. Since air-cooling appears to be a quite promising method for turbojet engines and more research information is available than for liquid-cooling, the discussion of methods of cooling is limited to that type.

Air-Cooling Methods

When air-cooling is incorporated in an engine, the entire engine design must be considered so that the air will be utilized most effectively. A possible engine configuration is shown in figure 9. Air is ducted from the discharge or one of the rear stages of the compressor to cool the turbine rotor and stator and is then discharged into the gas stream downstream of the turbine to provide additional thrust. A hollow turbine shaft provides a very convenient method of ducting cooling air to the rotor and can often eliminate many air-sealing problems.

Three methods of air-cooling are shown in figure 10. The most conventional method used in all heat-transfer processes is convection cooling (fig. 10(a)). With this method, it is desirable to increase heat-transfer surface area on the heat-rejection side of the apparatus such as in the form of the fins shown. This method of cooling has been successfully used on air-cooled piston engines for many years. The method of cooling shown in figure 10(b) is less well known. A film of cool air is introduced through slots to form an insulating layer between the hot gases and the cooled surface. The thermal conductivity of air is very low, so that it is a good insulation medium, but the effectiveness of the layer of air is lost some distance downstream of the slot when it mixes with the hot gases. This disadvantage is eliminated by transpiration cooling (fig. 10(c)), because air is continuously bled through the entire area of a porous surface. Transpiration cooling is the most effective method of air-cooling known at the present time. A comparison of the cooling effectiveness of these three methods of cooling is given in reference 9.

Air-cooled turbine blades of about 2-inch chord utilizing these various methods of cooling are illustrated in figure 11. The hollow blade was used by the Germans in some of their engines in 1945. A survey of their work on cooling turbojet engines and turbosuperchargers is given in references 10 to 12. The cooling effectiveness of the hollow blade is so low that excessive quantities of cooling air are required; consequently, efforts were made to provide added internal heat-transfer surface area. The tube-filled blade was an early attempt of the NACA to provide this extra surface area. Some results of tests on and methods of manufacturing this type of blade are given in references 2, 3, and 13 to 17. Although more recent blade developments have led to superior blade configurations so that the tube-filled blade configuration is presently of little interest, much valuable information has been obtained from this configuration: (a) The feasibility of building blades of nonstrategic materials was demonstrated (refs. 2, 3, 15, and 16); (b) suitable coatings for nonstrategic materials were investigated (ref. 17); (c) methods of blade fabrication including forming, brazing, heat-treatment, and suitable types of structure were studied (refs. 2 and 15); and (d) a relation was obtained between bar-stock stress-rupture life and blade life for nonstrategic blades made of 17-22A(S) steel (ref. 3).

The British (ref. 18) have used a somewhat different method of approach to the problem of adding internal surface area. Instead of packing a hollow shell with tubes and brazing the assembly together, they drilled holes in a solid blade to provide coolant passages with a greater internal surface area than is possible in a hollow blade.

Cooling of the leading and trailing edges is often difficult with tube-filled blades. In an attempt to improve cooling effectiveness in these regions, film cooling was investigated on the type of blade shown in figure 11(c), with the results reported in references 19 to 21. Cooling of the leading and trailing edges was effective, but blade durability was a serious problem (ref. 16). Research was conducted in Germany on blades having film cooling around the complete periphery (ref. 22), and similar blades have been built in this country. Although cooling is adequate for some cases, durability is usually poor. Another solution to the problem of leading- and trailing-edge cooling is to increase the thermal conductivity of the blade shell; consequently, shells that were copper-clad on the inside surface (fig. 11(d)) were investigated in reference 20. This type of structure is similar to that of copper-clad kitchen utensils in which the copper spreads the heat over the entire area of the utensil. The biggest disadvantage of the copper-clad blade is the weight increase, which raises the stress so much that the gains from cooling are practically eliminated.

A practical type of shell-supported convection-cooled blade construction is the corrugated-insert blade (fig. 11(e)). Large amounts of heat-transfer surface area can be added in the form of corrugated fins, and the fins can be made to extend well into the leading and trailing edges of the blade to ensure adequate cooling in these regions. An island is usually provided in the middle of the passage so that the corrugations can be of uniform amplitude. This island is blocked off from the cooling air. The island can be eliminated in small blades, but the corrugations will not be of uniform amplitude. Temperature data obtained from a corrugated blade are discussed in the next section.

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In all the turbine blades discussed up to this point, the blade shell has been the primary support member for carrying the stresses due to centrifugal forces. Since the shell is exposed to the gas stream, it is also the hottest member of the blade; and, therefore, its stress-carrying capacity is lower than that of cooler portions of the blade. For this reason, blades have been designed and tested with the main stress-carrying member, or strut, submerged inside the coolant passage and operating at a lower temperature than the blade shell (figs. 11(f) and 12). The shell can be made thin and can be completely supported by the strut. In this manner the stresses in the shell are greatly reduced, and it can operate at higher temperatures. With higher shell temperatures, the heat transfer from the gas to the blade is reduced and the quantity of cooling air required is reduced. This type of blade shows great promise for future application in air-cooled turbine engines.

The strut-supported blade shown in figure 12 and investigated in reference 23 is an example of only one of several possible methods of construction. The blade shown can be fabricated by machining the component parts of the strut and base and then brazing the final assembly together. The shell can be attached by brazing or spot-welding. Research is currently being conducted on a strut-blade configuration having the strut cast in an integral piece. The final configuration is approximately the same as the one shown in the assembled view in figure 12.

The blade in figure 11(g) is a transpiration-cooled blade. The porous shell could be made from several materials, the most probable being woven wire cloth or porous sintered materials made from powdered metal. Only a limited amount of experimental data is presently available for transpiration-cooled turbine blades. Some results are given in references 24 to 27. Advantages and problems in the use of transpiration cooling are discussed in reference 28. Reference 29 states that, if transpiration-cooled blades are to operate satisfactorily over wide ranges of flight altitude and flight Mach number, the cooling air must be metered to local positions on the blade surface. This can be accomplished by incorporating orifices in the turbine blade base as shown in figure 13. In addition to providing more uniform cooling over the wide range of operating conditions, this method of fabrication greatly simplifies blade fabrication, because metering the air at the orifices instead of through the porous surface permits fabrication of blades with uniform permeability around the blade periphery. The strut shown in figure 13 has the dual purpose of dividing the blade into compartments so that an orifice can meter air to local positions on the blade surface and providing a support member for the porous shell. Porous sintered shells usually require an internal support because of low shell strength, and woven wire shells need the support to provide rigidity.

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Experimental Temperature Data

Experimentally measured turbine blade temperatures are presented in figure 14 for the corrugated-insert blade. The coolant-flow ratio used as the abscissa is defined as the ratio of the air used for turbine-cooling purposes to the total flow of air through the compressor. For the uncooled condition the blade temperature is over 200° F lower than the turbine-inlet temperature, because the gas total temperature relative to the turbine rotor blades is less than the gas total temperature relative to the stator blades, as a result of high rotative speeds of the turbine and high gas velocities at the stator exit. The use of only 2 percent of the compressor air for turbine rotor cooling will reduce the blade temperature over 400° F below that of an uncooled blade, or approximately 650° F below the turbine-inlet temperature. These results show the substantial blade temperature reductions possible with very nominal amounts of cooling air. In addition, they show the success obtained in developing methods of predicting the average blade temperature of this type of blade and lend encouragement to using these methods for predicting cooling requirements for other conditions. The variation between measured and predicted temperatures is less than 35° F. The predictions are based essentially on the methods discussed in references 30 and 31.

A more complete summary of methods of analytically predicting air-cooled blade temperatures is presented in reference 7. In addition, a method is presented in reference 32 that provides a quick rough evaluation of the cooling effectiveness and pressure-drop characteristics of convection-cooled turbine blades. Methods are presented in reference 33 for rapidly evaluating the heat-transfer and pressure-drop characteristics of corrugated-insert air-cooled blades.

A comparison of the cooling effectiveness of a corrugated-insert blade with that of a strut-supported blade is given in figure 15. The temperature comparison is based on the temperatures of the stress-carrying members, which are the shell of the corrugated blade and the strut of the strut-supported blade. The data for the strut blade were obtained from reference 23. Figure 15 shows that the coolant flow required for the strut-supported blade is about half that required for the corrugated blade in order to obtain a specified blade temperature. At the very low flow rates, this difference will have very little effect on the over-all engine performance; but the results of figures 7 and 8 show that, as the cooling load becomes more severe, the savings in cooling air with the strut-supported blade could result in appreciable gains in engine performance. Experimental and calculated temperatures for the strut-supported blades are also compared in figure 15. Again the agreement is very good. The calculated temperatures were obtained by the methods described in reference 34.

Caution must be observed in generalizing the results obtained from research on the blades shown in figures 11 to 15. All these blades had a chord of approximately 2 inches or more and would be suitable for many turbojet engines. For some high-compressor-pressure-ratio turbojet engines and for most turboprop engines, however, the blades are much smaller. Research is presently being conducted on these smaller blades, and also on blades that have such thin cross sections that use of internal heat-transfer surface area is difficult. Preliminary studies indicate that scaling down some of the cooling schemes such as the strut blade appears to be feasible, but in addition new types of design will be required for many applications. The research has not advanced far enough to warrant discussion.

Air-cooled blade durability is equally as important as the blade temperature reduction that is possible by cooling. Some of the endurance investigations, conducted mostly on nonstrategic tube-filled blades having a critical blade section stress of about 23,500 pounds per square inch, are reported in references 2, 3, 16, and 17. Air-cooled turbine blade life of 350 hours at full-power engine conditions has been obtained with no indication of failure (ref. 3). Although further endurance programs are required on other types of blades, other blade materials, and blades operating at higher stress levels to obtain conclusive results, it is indicated that, with proper design, air-cooled turbine blade life should be satisfactory. Turbine blade stress levels in excess of 40,000 pounds per square inch are being investigated, and it is indicated that blade stresses as high as 50,000 pounds per square inch may be feasible.

Blade Pressure Losses

In addition to the heat-transfer characteristics of turbine blades, the pressure-loss characteristics are important, because they determine to a large extent the flow capacity of the blades and the point at which cooling air should be bled from the compressor. There is a general relation between heat transfer and friction, so that blades with high cooling effectiveness also have relatively high pressure losses. A quick method of calculating the pressure changes through air-cooled

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turbine blades is presented in reference 35. Friction factors for two air-cooled turbine blade configurations were measured and reported in reference 36, in which air-cooled blade pressure losses were calculated within ±6 percent.

The general practice in the design of air-cooled turbine blades has been to design for a maximum cooling-air Mach number of 0.5 at the blade tip. At a given inlet supply pressure, the flow at a Mach number of 0.5 is about 75 percent of the choked flow. It is felt that designing for higher Mach numbers leaves little margin of safety for extreme conditions that may be encountered in engine operation. A method of blade design that takes into account both pressure loss and heat transfer is presented in reference 33 for corrugated-insert blades.

AIR-COOLED TURBINE DISK CONFIGURATIONS

The use of air-cooled turbine blades will require a type of turbine rotor construction different from that in current use. There is, however, a considerable amount of freedom in the type of design possible. Two main types of turbine rotor are the split disk (fig. 16) and the shrouded disk (fig. 17). With either construction the cooling air can be supplied from the upstream direction, the downstream direction, or through a hollow turbine shaft. With any of these possible types of construction, internal vanes are required in the turbine rotor to direct and help pump the cooling air out to the blades. Vane configurations for two experimental split-disk air-cooled turbines are shown in figure 18. In one case the vanes were curved to provide an inducer section for the cooling air, while in the other case straight vanes were used. Experimental tests reported in reference 37 do not indicate the superiority of either type of vane construction, and in each case the pressure rise in the turbine disks is small. In general, the pressure supplied at the rotor hub will have to be as high as the static gas pressure at the turbine blade tips, or higher.

Up to the present time, experimental tests have been conducted on several turbines with the type of disk configuration shown in figure 16(b), and some of the results presented in references 38 to 40 indicate that disk cooling will be adequate with the amount of air required for blade cooling. Experimental results have not been obtained, however, at turbine-inlet temperatures in excess of 1900° F. In the experimental turbines the downstream inlet was required in order to minimize the alterations to a commercial engine when incorporating turbine cooling. A discussion of some of the relative merits of other types of disk construction is given in reference 41.

PROBABLE FUTURE USE OF VARIOUS TURBINE-COOLING METHODS

Most experimental research on turbine cooling has been concerned with investigating various possible types of blade configurations to determine cooling effectiveness, fabrication problems, and expected durability. Investigations have been conducted in commercial engines modified to accommodate air-cooled turbines, and the turbines were usually made of nonstrategic materials. A combination of this experimental research and analysis has made possible the verification of analytical procedures, and cycle calculations have been made to determine the areas of operation in which future use of turbine cooling will 3511
be most profitable. The use of cooling is particularly promising for turbojet engines powering aircraft at supersonic speeds and for turboprop engines or other shaft-power turbine engines used in subsonic and transonic flight and stationary or marine power plants.

The relative merits of three methods of blade cooling (convection, film, and transpiration) for future use in cooled turbojet engines are indicated in figure 19. The relative coolant flows required for these various types of cooling are shown for turbine-inlet temperatures of 2000° and 3000° F and for flight Mach numbers of zero and 2.5. The results can be used to indicate trends only, because they are not the result of a design study. No consideration is given to cooling-air pressures that may be required or to whether actual blades will be feasible for the various conditions based on size, durability, and so forth. The calculations are based on an assumed blade temperature of 1250° F, compressor-discharge air bleed with no refrigeration of the cooling air, and results presented in reference 9. It was assumed that the compressor had a pressure ratio of 10 at a Mach number of zero and a pressure ratio of 6 at a Mach number of 2.5. It should be remembered that no experimental information is available concerning cooling at a gas temperature as high as 3000° F; consequently, results at 3000° F are based on the extrapolation of correlation methods obtained at lower temperature levels. The results shown at this temperature, therefore, must be considered approximate.

At low flight speeds figure 19 shows that the quantity of coolant required for turbine-inlet temperatures up to 2000° F (almost 400° F above current practice) is relatively small for either convection or transpiration cooling. Film cooling appears to be impractical mainly because of the problem of blade durability; but, in addition, the flow requirements are higher than for other methods. The convection-cooling bar is representative of the better shell-supported blades (see fig. 11(e)). Strut-supported-blade cooling-air requirements would be intermediate between those shown for convection and transpiration cooling.

As turbine-inlet temperature is increased to 3000° F at low flight speeds, the cooling requirements become more severe, and film cooling appears to be completely out of the question. Convection cooling is possible, but relative to transpiration cooling the requirements are high.

As flight speeds are increased, the ram-air temperature increases considerably, so that the temperature of the compressor-discharge air used for turbine cooling rises rapidly. The high cooling-air temperature makes turbine cooling more of a problem at high flight speeds. As shown in figure 19, at a flight Mach number of 2.5 the blade cooling problems are just as severe for a turbine-inlet temperature of 2000° F as they are at low flight speeds for a turbine-inlet temperature of 3000° F. Increasing the turbine-inlet temperature to 3000° F at a flight Mach number of 2.5 and using compressor-discharge air create a cooling problem so severe that transpiration cooling is the only air-cooling method presently known that could operate efficiently. It is possible, however, to provide a certain amount of refrigeration to the turbine-cooling air. Several systems including refrigeration for utilizing compressor air for turbine cooling at high flight Mach numbers are presented in reference 42. With refrigeration systems, convection cooling at high flight speeds and high temperatures is also feasible. At flight Mach numbers up to somewhat over 2.0, cooling-air refrigeration is probably unnecessary.

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It can be concluded from this study and other studies that, for gas temperatures up to about 2500° F and flight Mach numbers up to at least 2.0, convection-cooled blades of the corrugated-insert and strut-supported types may be adequate. At higher temperatures and higher flight speeds, transpiration cooling may be required. The cooling of small or thin blades appears more difficult, and maximum temperature and flight speeds may be lower than for the cooling methods discussed.

There is also the question concerning the relative merits of air- and liquid-cooling. Because of its heat-transfer properties, water is one of the best liquid coolants and will probably find use in almost any type of liquid-cooling system. In most cases the heat picked up by the water will be rejected to ram air. At Mach numbers in excess of 2.0 in the stratosphere, the ram-air temperature is higher than the boiling point of water at reasonable pressures in the radiator. With water-cooled systems, therefore, the flight Mach number will be limited to approximately 2, or else systems will have to be designed that are capable of operating at extremely high water pressures - preferably above the critical pressure of water (3206 psi). Such high pressures would suggest the use of a rotating heat exchanger to eliminate the necessity of transferring the high-pressure water through seals from rotating to stationary parts of the engine.

Another problem associated with water-cooling systems in military aircraft is the vulnerability to enemy attack. Thus, reliability is somewhat more questionable than for air-cooled systems. A more complete description of several methods for and problems involved in dissipating the heat in a liquid-cooled turbine is given in reference 43. Cooling losses are always less (neglecting effects of radiator drag) with water-cooling than with air-cooling, but for turbojet engines the difference in performance probably is not of very great importance. Therefore, it is expected that air-cooling will find more use in turbojet applications. For very small turbojet-engine turbine blades it is possible, however, that better cooling could be obtained with liquids than with air.

For turboprop applications, air-cooling losses are larger than for turbojets, but substantial performance improvements are possible with either air- or liquid-cooling to permit higher turbine-inlet temperatures. In addition, the flight Mach numbers for turboprop aircraft will probably always be subsonic or transonic, so that heat rejection with liquid-cooling systems will not be a serious problem. For aircraft application, the better performance of liquid-cooled systems will have to be weighed against lower vulnerability of the air-cooled systems. Either type of cooling system can probably be utilized satisfactorily. For stationary or marine power plants, however, there appear to be no particular advantages of air-cooling over liquid-cooling as a method to permit high-temperature operation; consequently, liquid-cooling will probably be more satisfactory, because it will result in lower fuel-consumption rates. For either air- or liquid-cooling, the potential gains in engine performance through the use of higher turbine-inlet temperatures for both shaft- and jet-power turbine engines are well worth the effort required to build turbine cooling into the engine design.

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CONCLUDING REMARKS

This review of the status, methods, and potentials of gas-turbine cooling can be summarized as follows:

Analytical Studies

1. The power output of nonafterburning turbojet engines and of turboprop engines can be increased by a factor of 2 or more by using turbine cooling to permit increases in the turbine-inlet temperature.

2. For afterburning turbojet engines, the effect of turbine cooling to permit increased turbine-inlet temperatures is primarily to decrease fuel consumption. In addition, for higher compressor pressure ratios combined with high turbine-inlet temperatures, large increases in thrust are obtainable at slightly improved specific fuel consumption.

3. Appreciable increases in turbine stress level are indicated through use of turbine cooling to reduce turbine metal temperature. Increased turbine stress levels allow reduction of turbine frontal area. The reduced metal temperatures also widen the choice of possible turbine materials, many of which contain only small amounts of critical alloying elements.

4. For nonafterburning turbojet engines operating at a flight Mach number of 2 in the stratosphere and constant turbine-inlet temperature of 2000° F, approximately a 1-percent loss in thrust accompanies each percent of air bled from the compressor exit for turbine cooling. The effect of turbine cooling on specific fuel consumption is negligible at most operating conditions for nonafterburning engines. The prevalent practice of bleeding air overboard for cabin cooling, accessory drives, and so forth results in thrust losses more than double the loss for turbine cooling, and in addition the specific fuel consumption increases over 2 percent for every percent of overboard bleed air.

5. For turboprop engines at subsonic speeds, the power reduction resulting from air-cooling at a constant turbine-inlet temperature of 2000° F is somewhat higher than for a turbojet engine, but the net gains in power resulting from higher turbine-inlet temperatures are still very large. In addition, operation at the higher turbine-inlet temperature can result in an actual decrease in specific fuel consumption, including the effects of cooling, relative to low-temperature uncooled engines.

6. Liquid-cooling has an extremely small effect on engine performance (neglecting radiator drag losses). In most cases, liquid-cooling will cause less than 1-percent loss in performance for turboprop engines.

7. Based on general heat-transfer analyses, it appears that convection-cooled corrugated-insert and strut-supported turbine blades will probably be satisfactory for most turbojet applications, except for operation under the combination of very high gas temperatures (2500° F) and high flight Mach numbers (above 2), where transpiration cooling may be required. For small or thin blades the maximum permissible temperatures and flight Mach numbers may be lower.

Experimental Studies

1. At present, the most promising types of air-cooled turbine blades for conventional turbojet engines are the corrugated-insert and strut-supported blades. Further research is required, and is under way, on both small and thin turbine blades. Preliminary studies indicate that scaling down some of the cooling schemes, such as the strut blade, appear to be feasible, but in addition new types of design will be required for many applications.

2. Analytical blade temperature predictions have been found to give reasonable agreement with experimentally measured temperatures for corrugated-insert and strut-supported turbine blades.

3. Endurance investigations have been conducted on a limited number of air-cooled turbine blades. Although further endurance programs are required on other types of blades, other blade materials, and blades operating at higher stress levels, in order to obtain conclusive results, it is indicated that with proper design air-cooled turbine blade life should be satisfactory.

4. Many types of air-cooled turbine disk designs appear to be feasible. Experimental results from several turbine disks of one type of design indicate that disk cooling will be adequate with the amount of air required for blade cooling. Experimental results have not been obtained, however, at turbine-inlet temperatures in excess of 1900° F. 3511

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, December 2, 1954.

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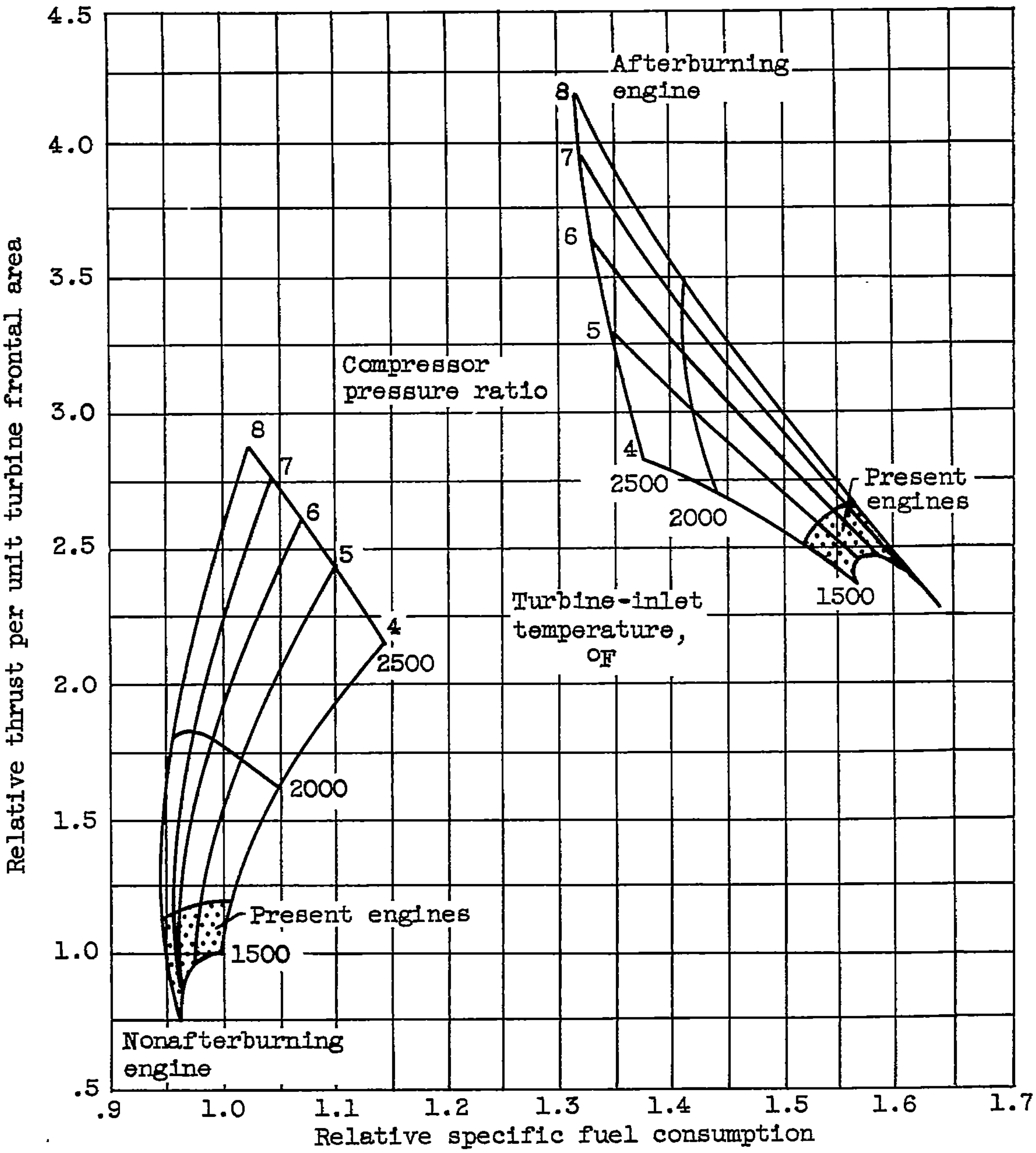

Figure 1. - Turbojet-engine performance for various turbine-inlet temperatures and compressor pressure ratios at flight Mach number of 2 in stratosphere (from ref. 1). Turbine hub-tip radius ratio, 0.75; afterburner gas temperature, 3000° F.

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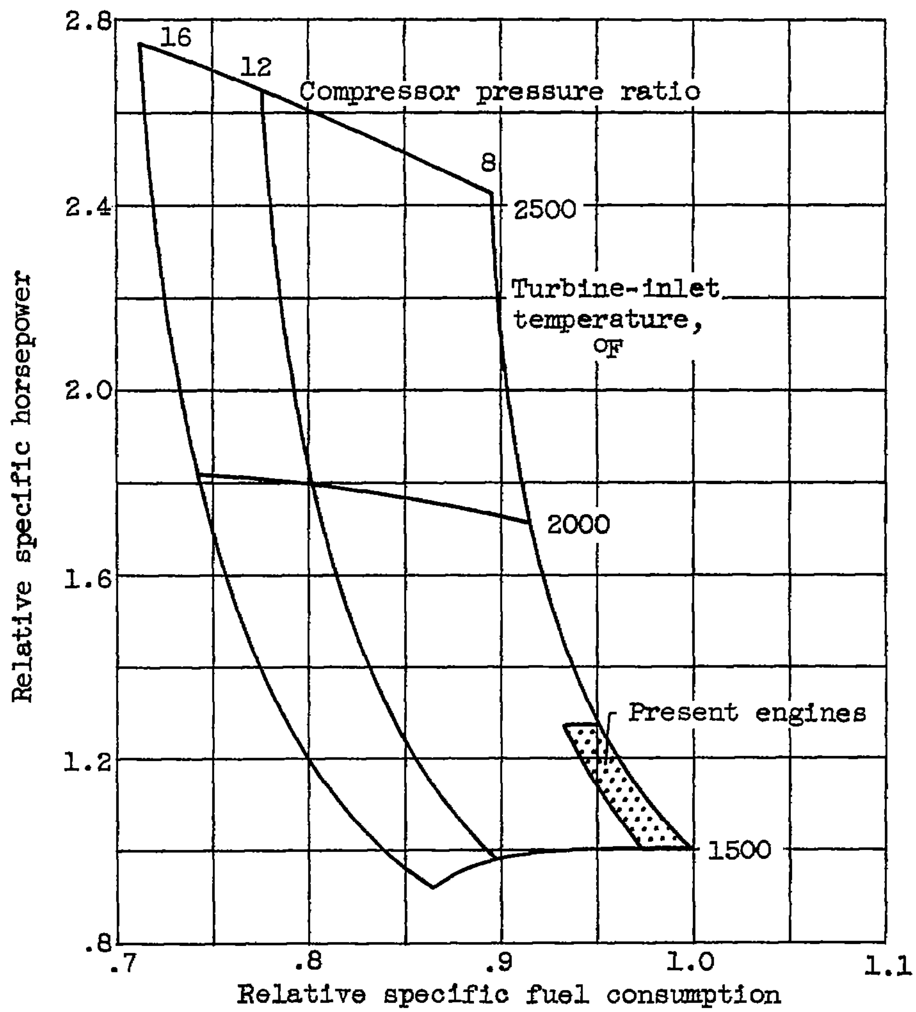

Figure 2. - Turboprop-engine performance for various turbine-inlet temperatures and compressor pressure ratios at sea-level static conditions.

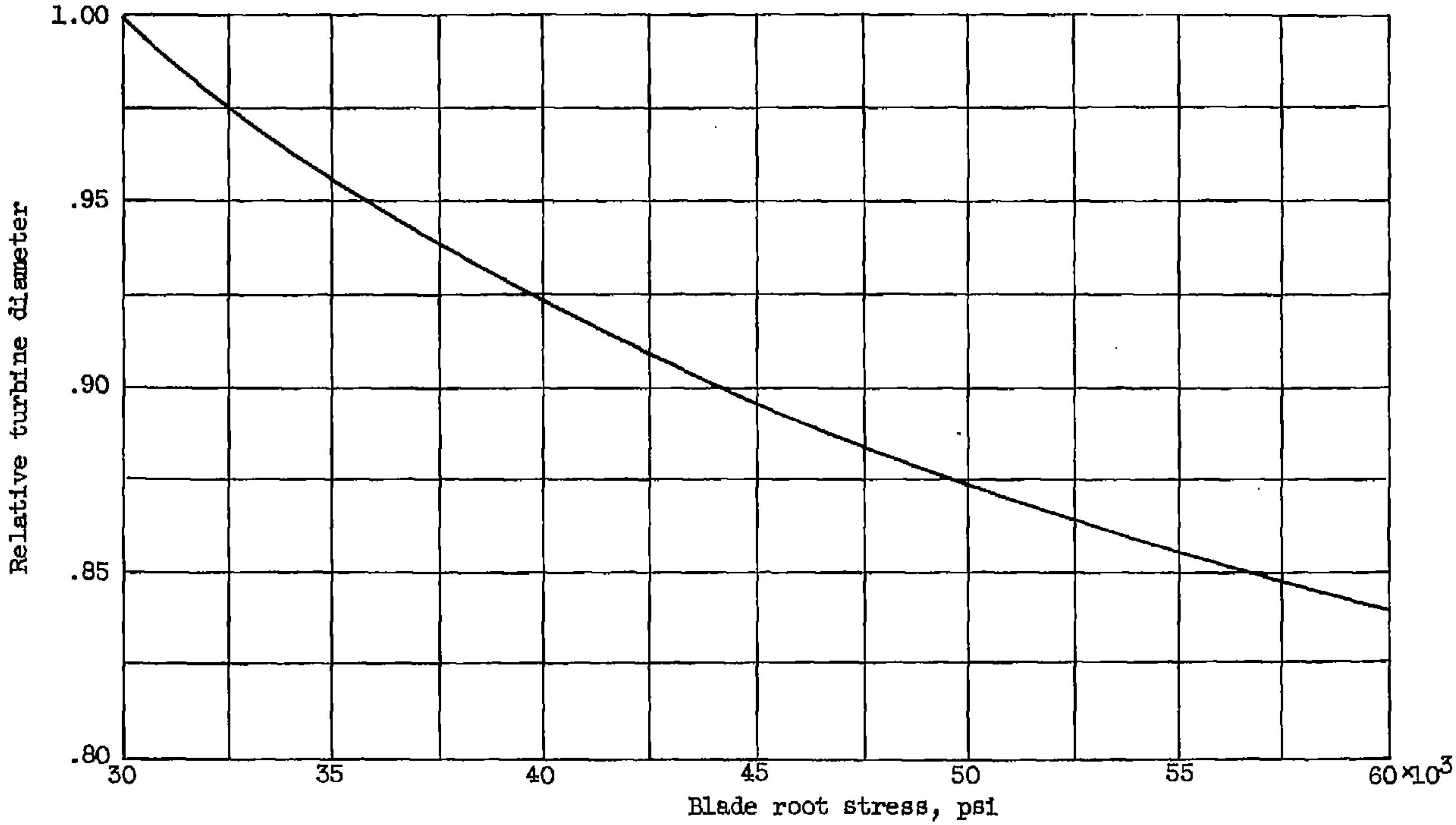

Figure 3. - Relation of turbine stresses to turbine diameter. High-output single-stage turbine for compressor pressure ratio of 7.

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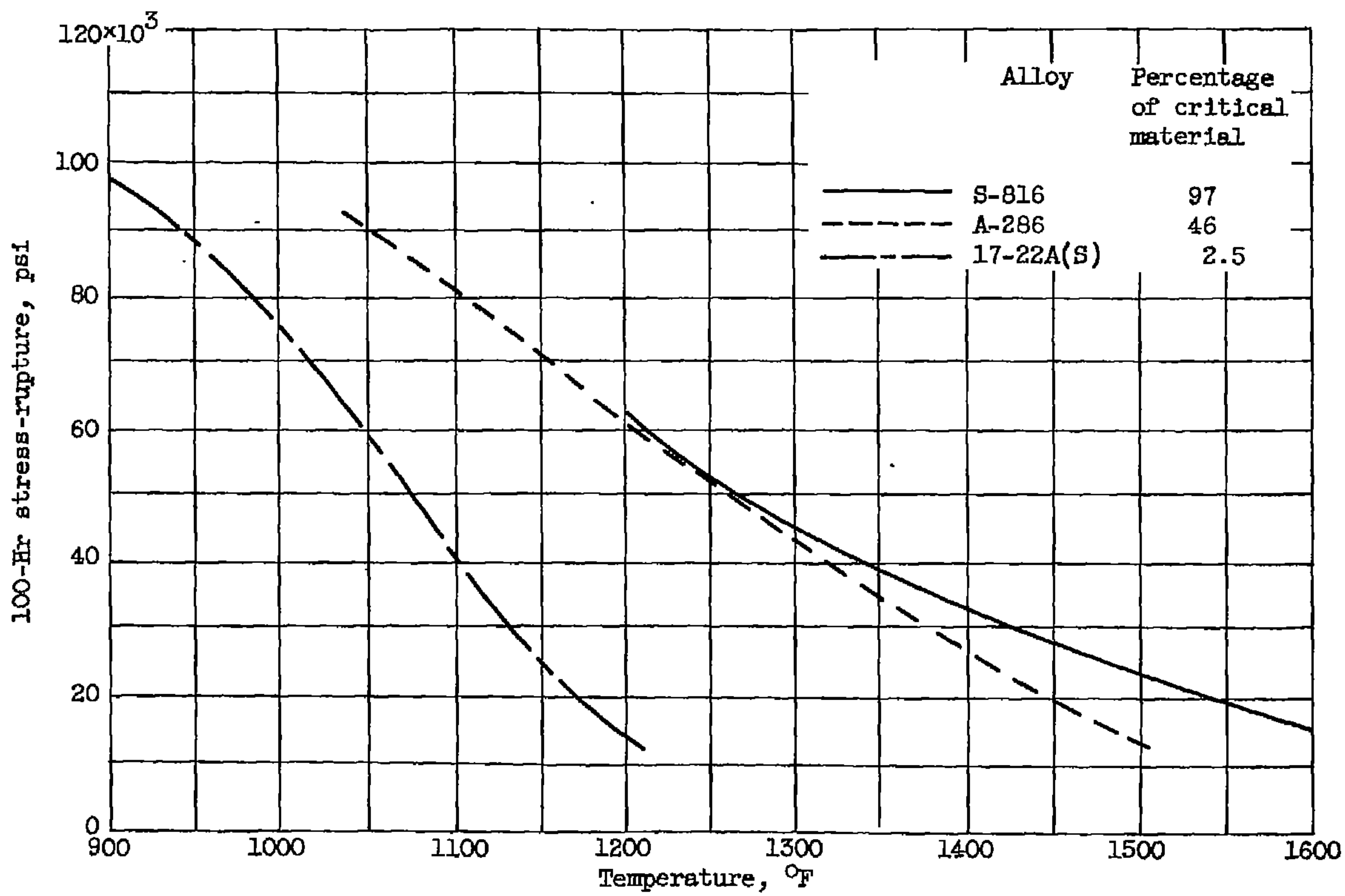

Figure 4. - Stress-rupture properties for possible air-cooled turbine blade materials.

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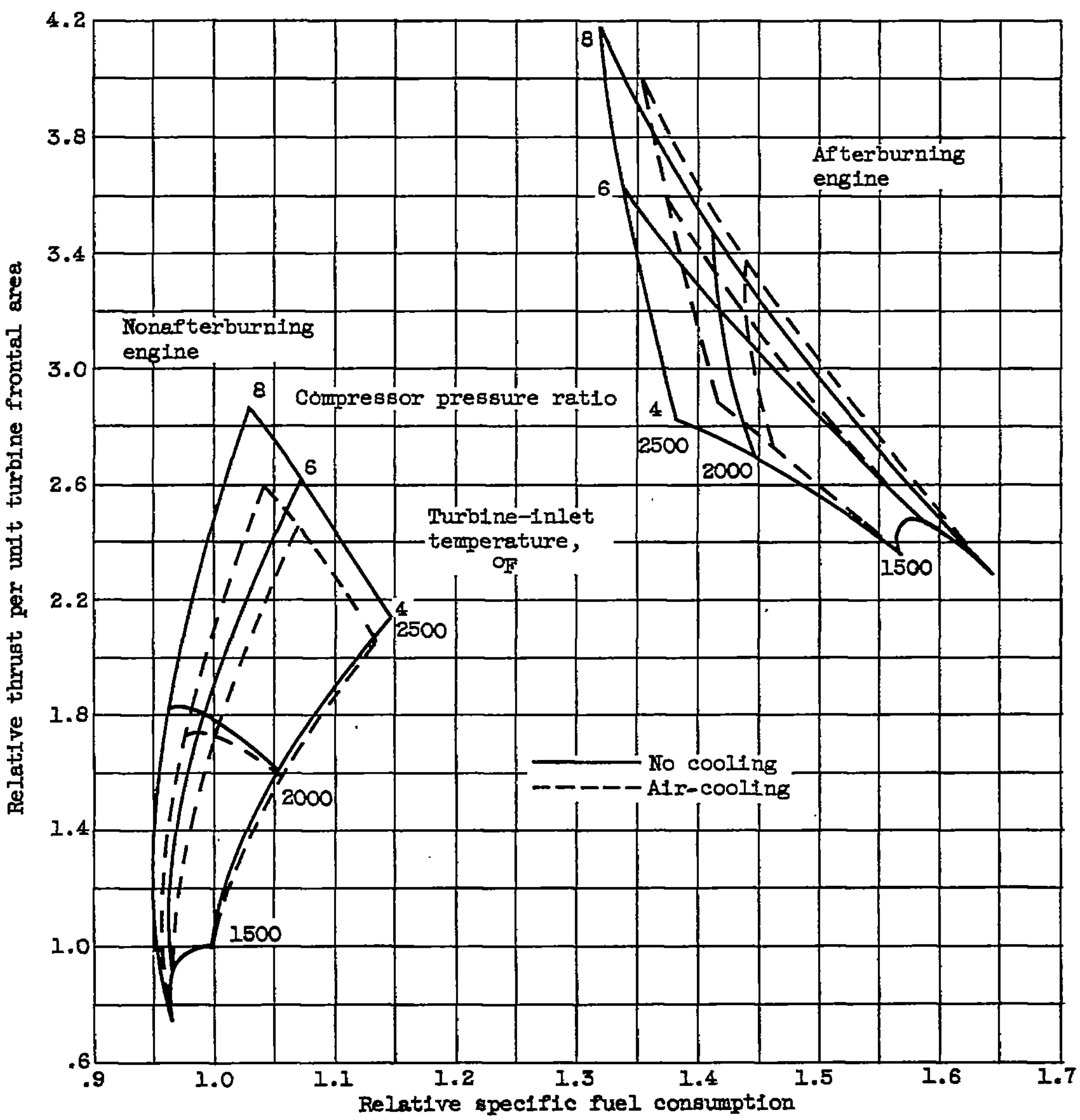

Figure 5. - Effect of air-cooling on turbojet-engine performance for various turbine-inlet temperatures and compressor pressure ratios at flight Mach number of 2 in stratosphere. Turbine hub-tip radius ratio, 0.75; afterburner gas temperature, 3000° F.

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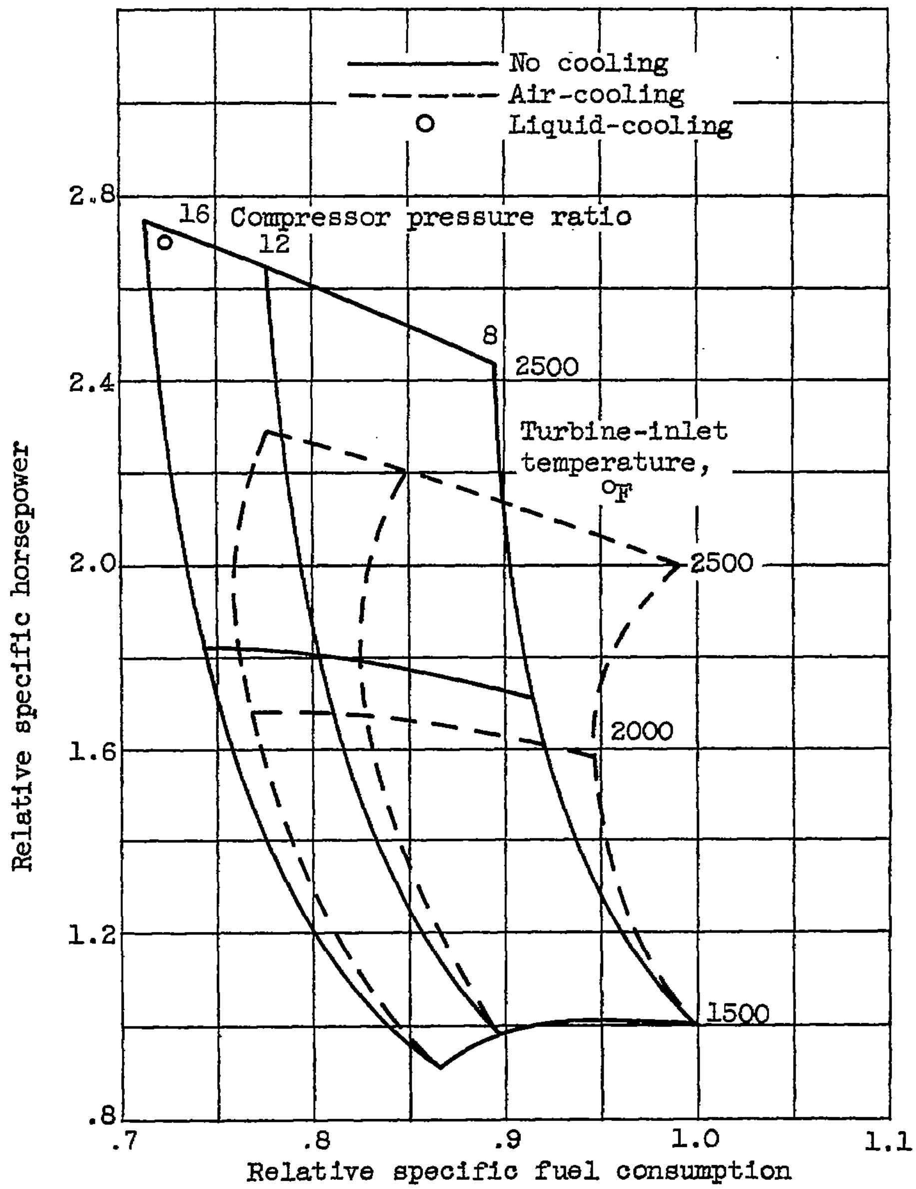

Figure 6. - Effect of air- and liquid-cooling on turboprop-engine performance for ranges of turbine-inlet temperature and compressor pressure ratio at sea-level static conditions.

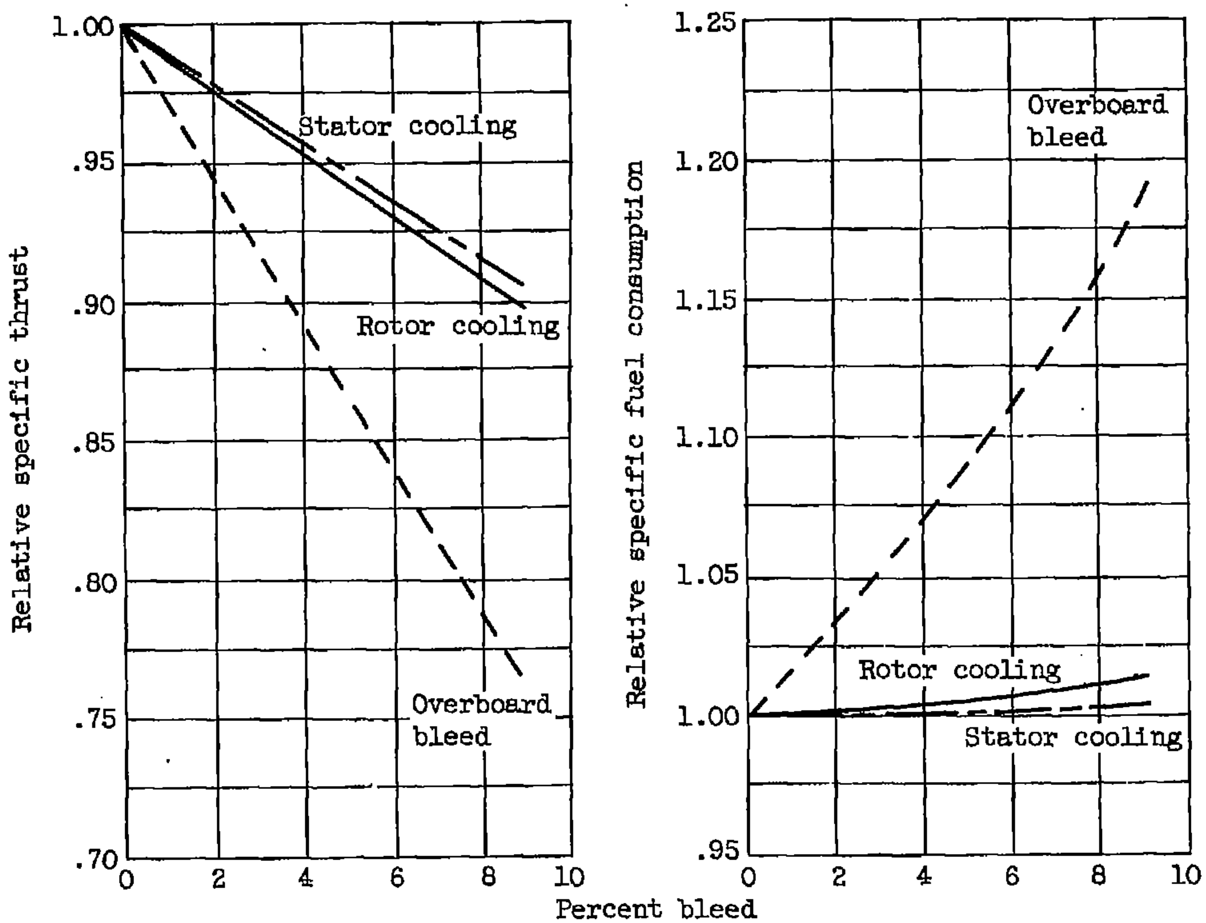

Figure 7. - Variation in performance of nonafterburning turbojet engine with various types of compressor-discharge air bleed. Turbine-inlet temperature, 2000° F; compressor pressure ratio, 6; flight Mach number of 2 in stratosphere.

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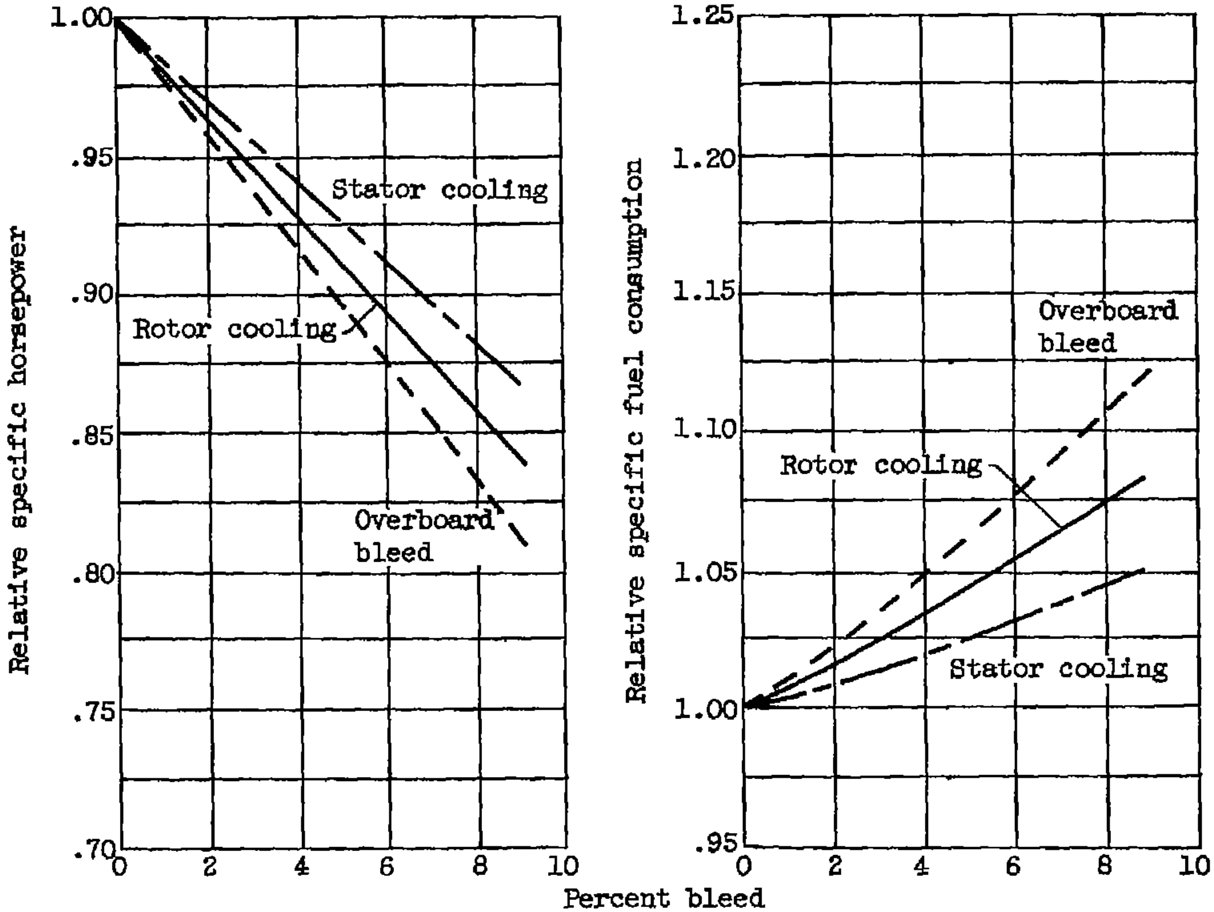

Figure 8. - Variation in turboprop-engine performance with various types of compressor-discharge air bleed. Turbine-inlet temperature, 2000° F; compressor pressure ratio, 12; sea-level static conditions.

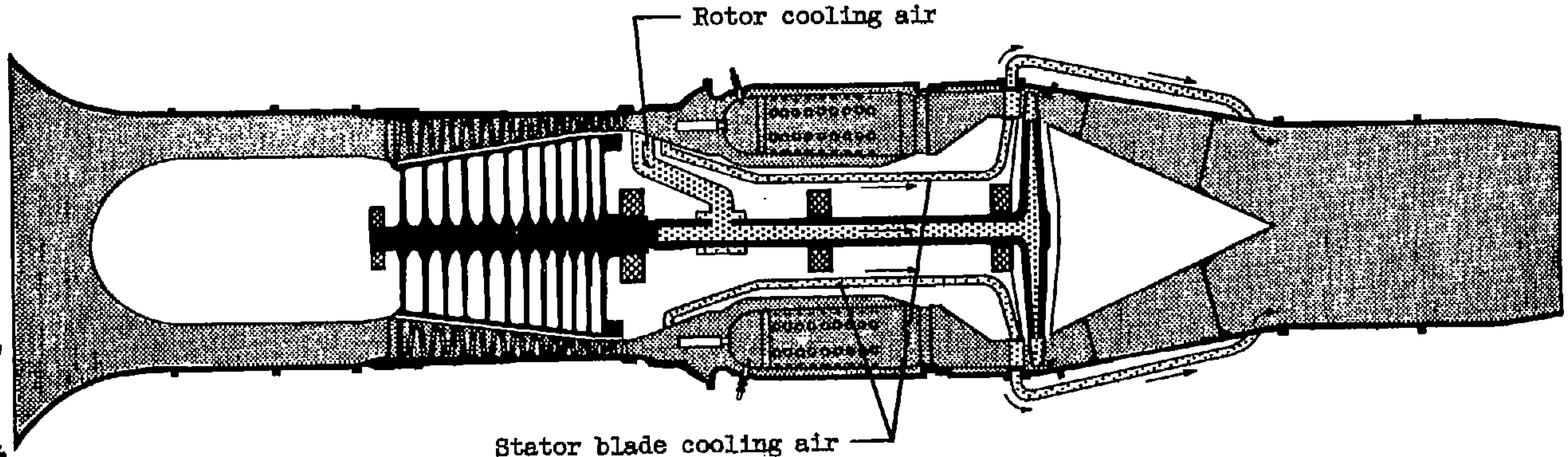

Figure 9. - Cross section of air-cooled turbojet engine.

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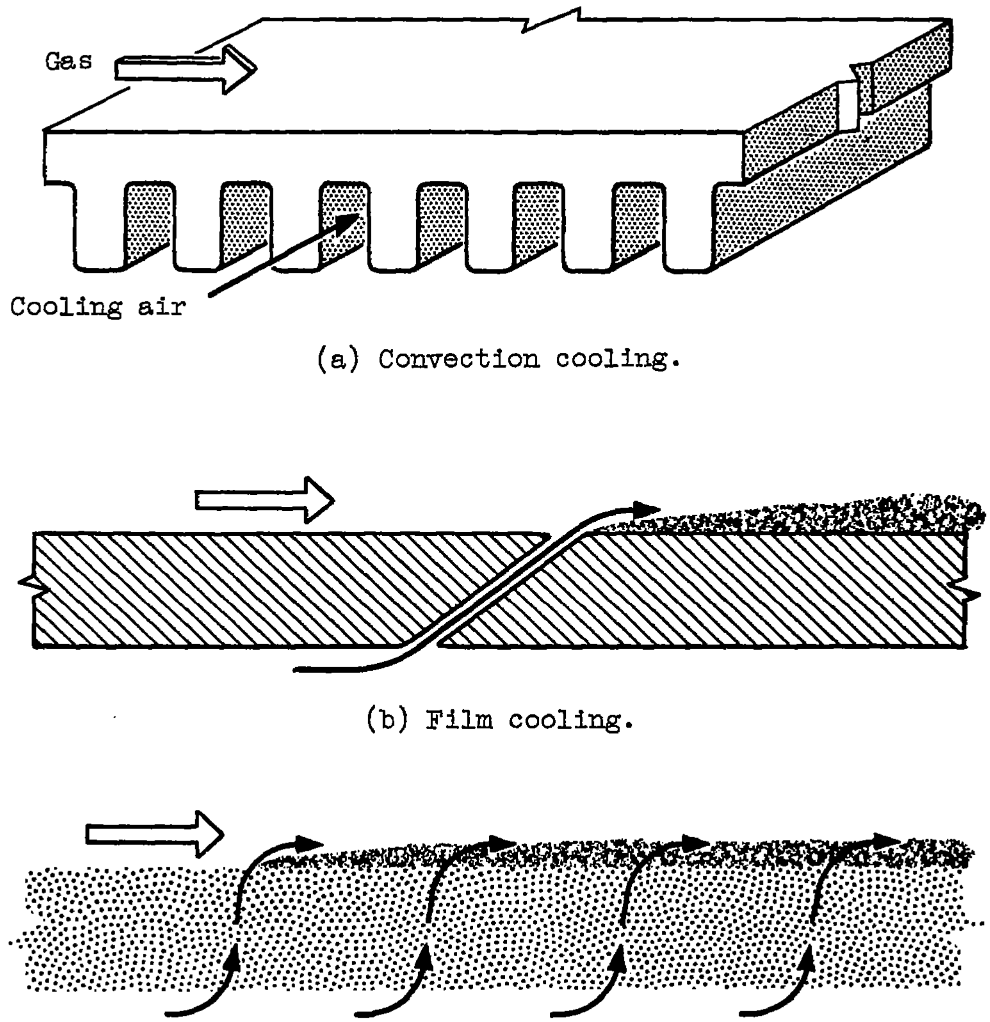

(a) Convection cooling.

(b) Film cooling.

(c) Transpiration cooling.

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Figure 10. - Three methods of air-cooling.

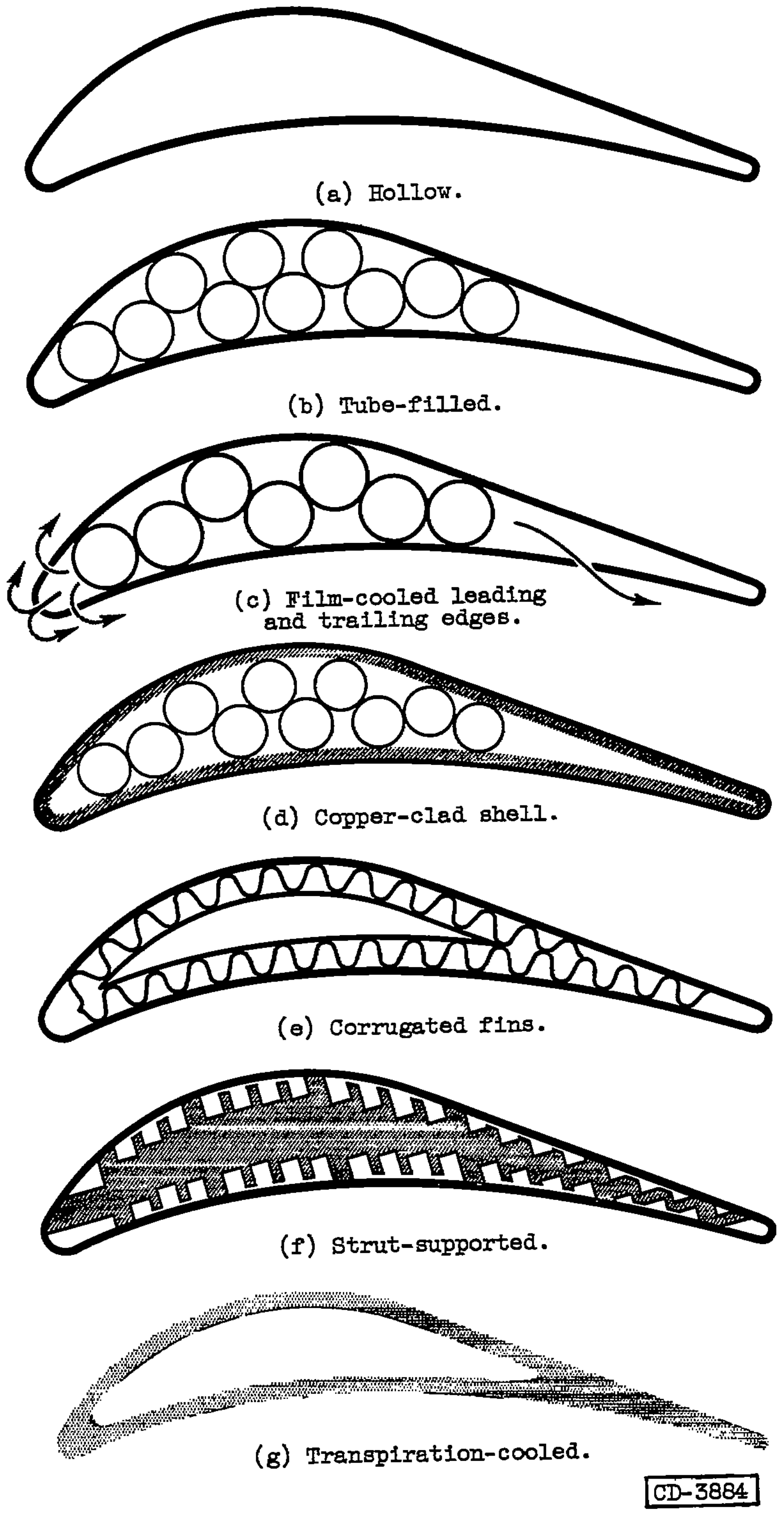

Figure 11. - Air-cooled turbine blade configurations.

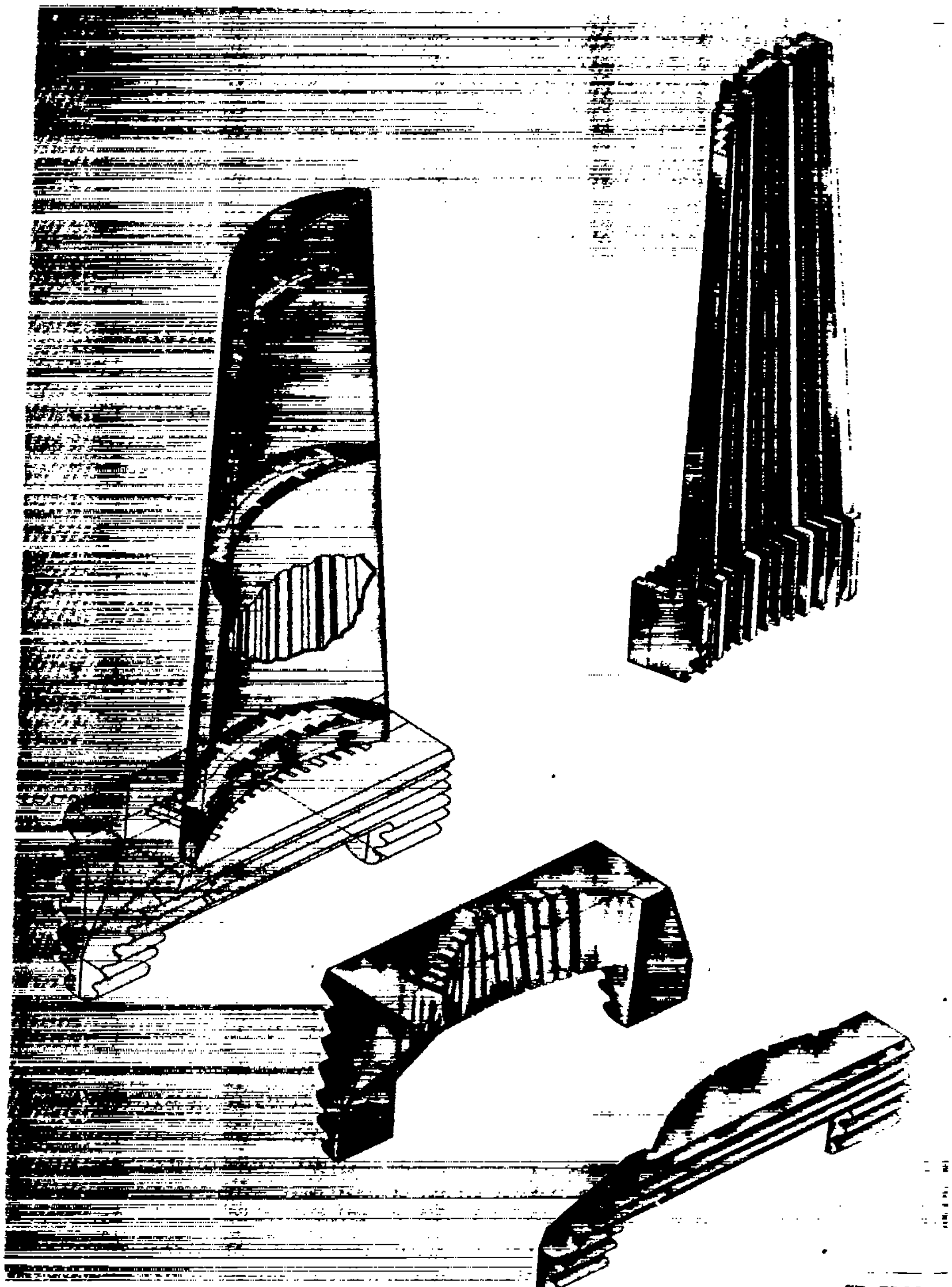

Figure 12. - Exploded view of air-cooled strut-supported turbine blade.

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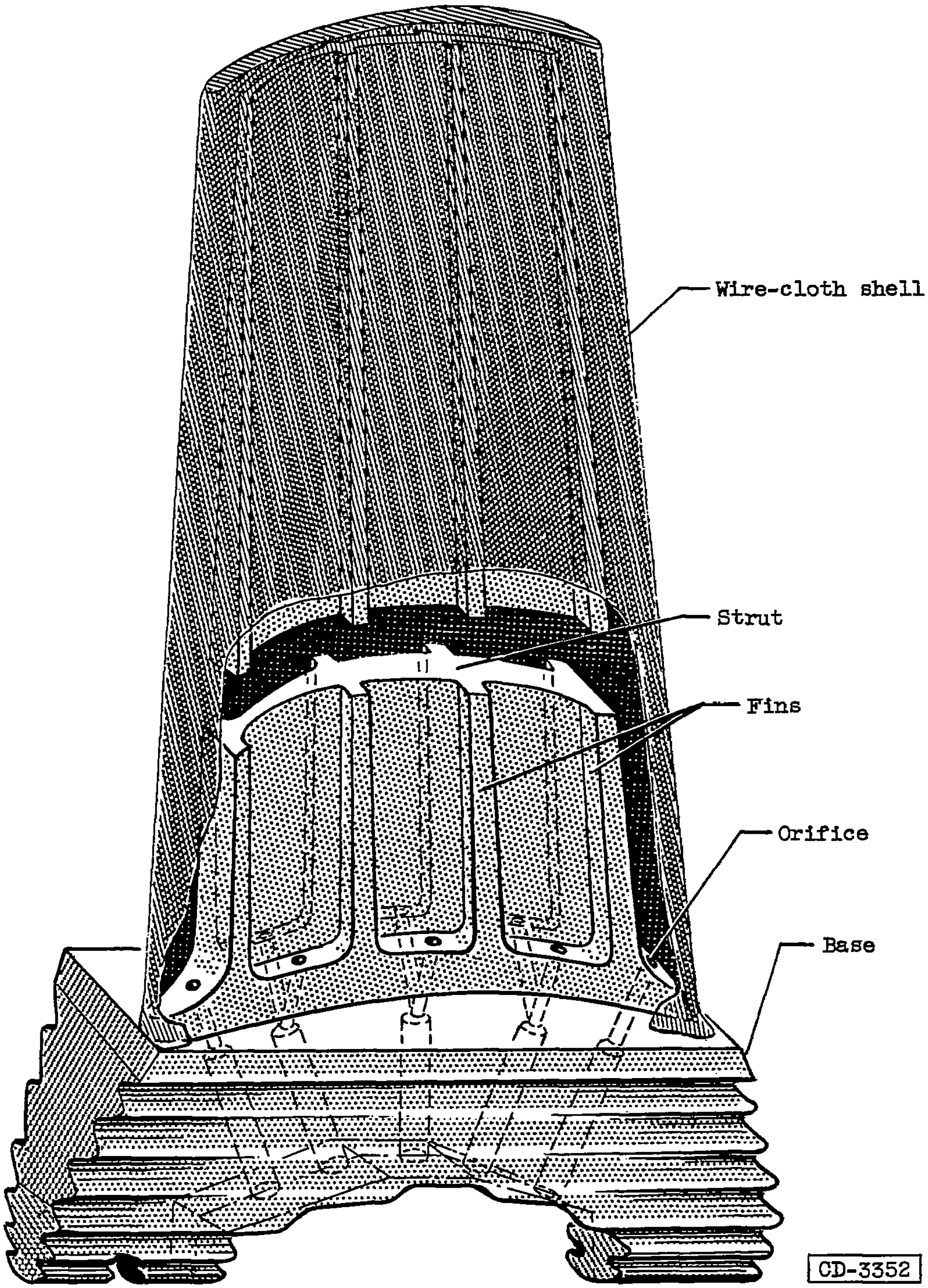

Figure 13. - Sketch of transpiration-cooled blade utilizing orifices in base for metering cooling air.

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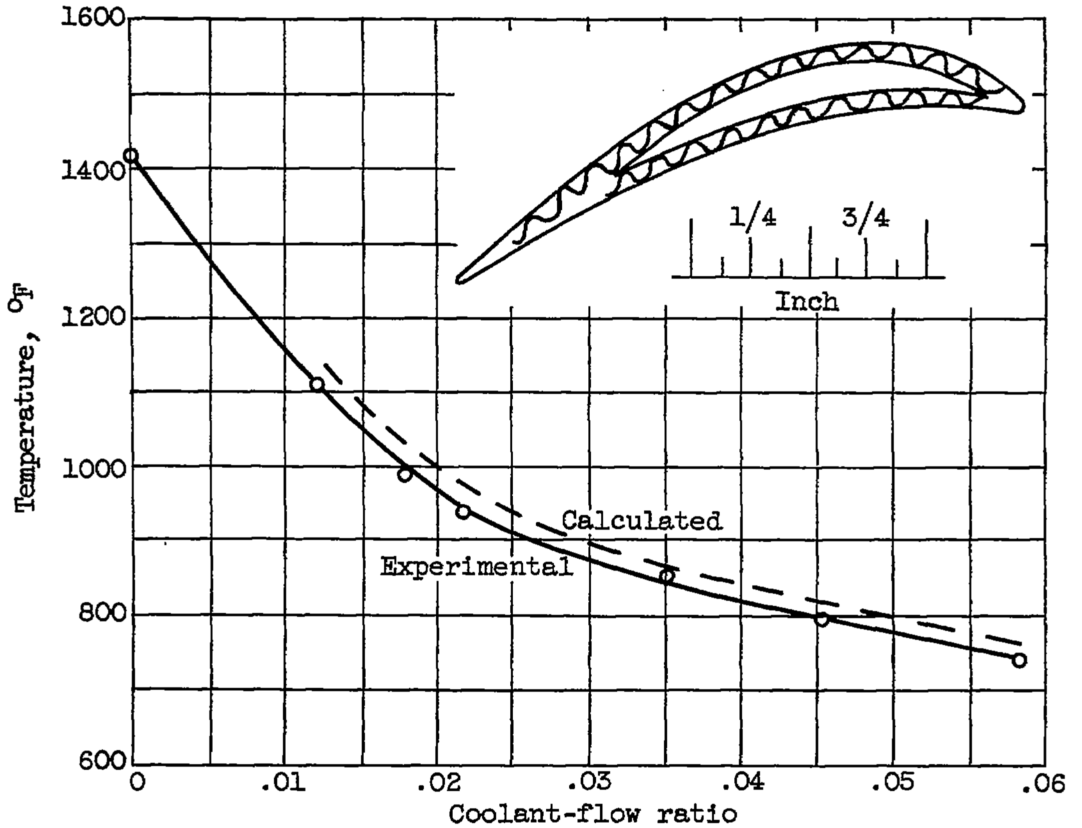

Figure 14. - Comparison of experimental and analytical blade temperatures. Turbine-inlet temperature, 1640° F; corrugated-insert blade.

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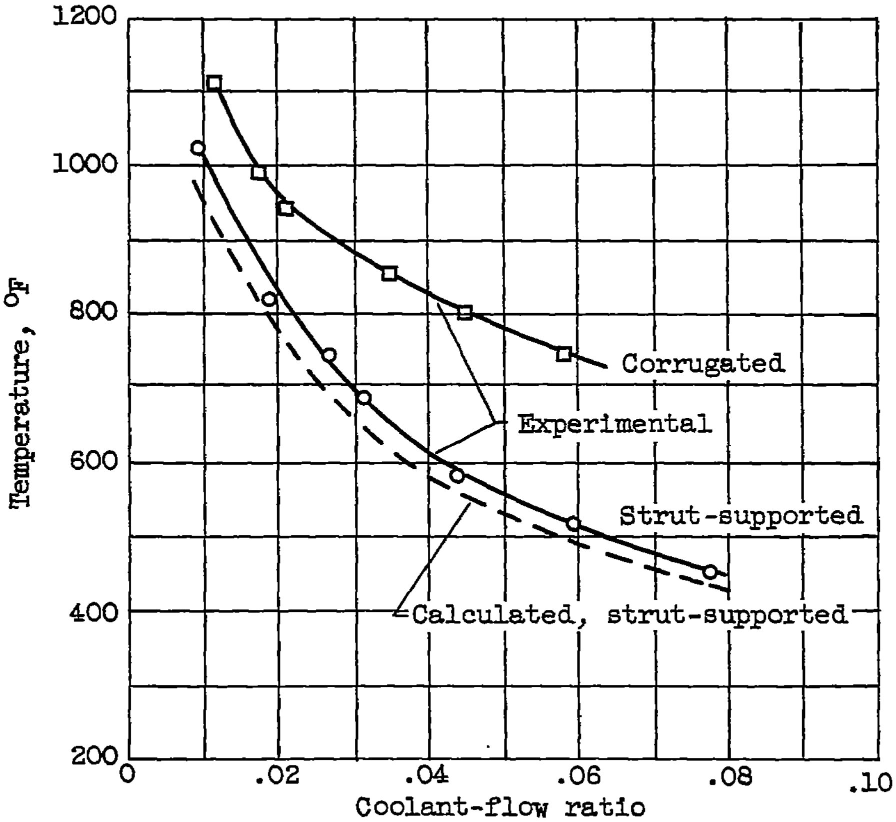
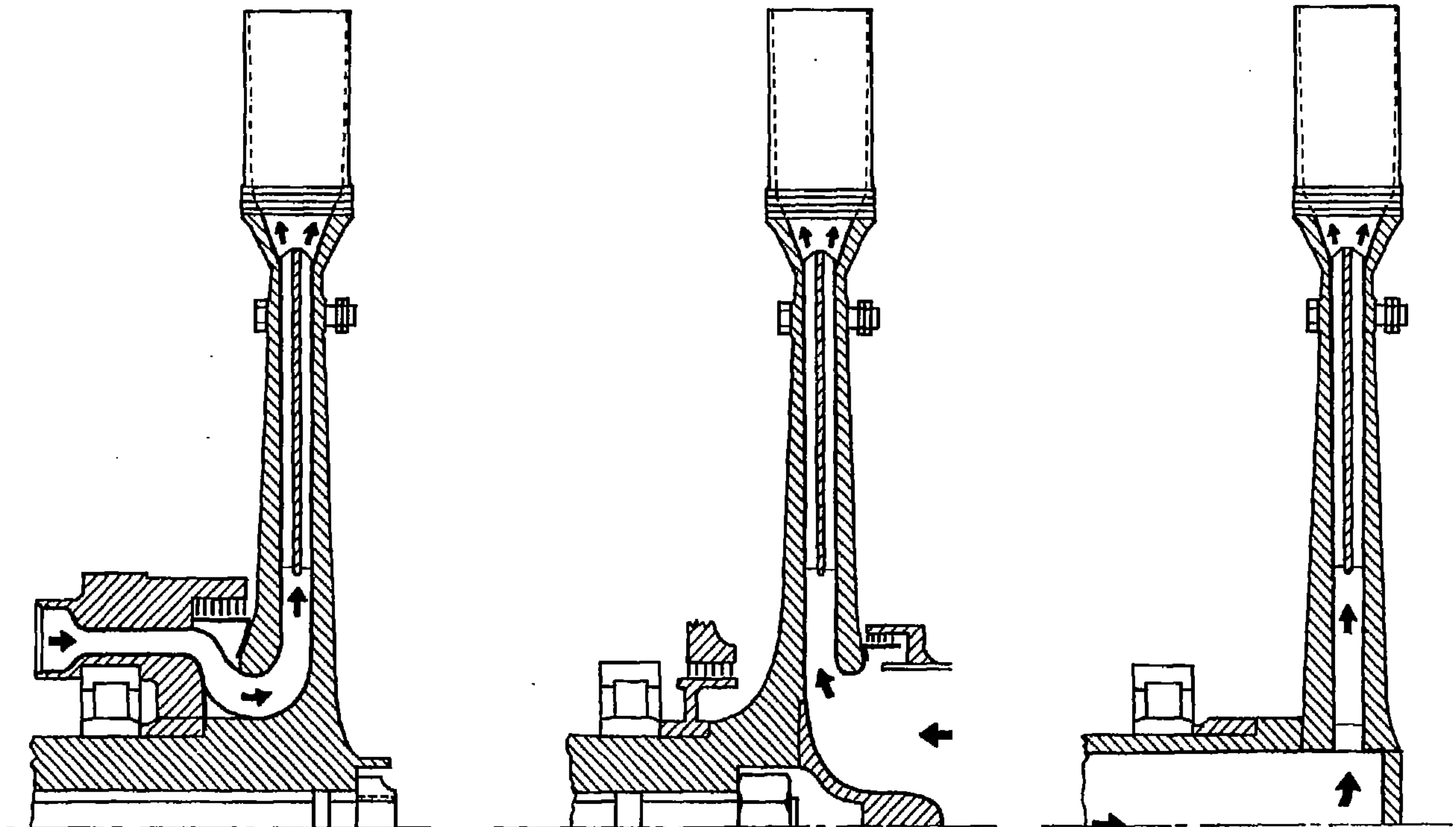


Figure 15. - Temperatures of two convection-cooled turbine blades. Turbine-inlet temperature, 1640° F.



(a) Upstream inlet.   (b) Downstream inlet.   (c) Hollow-shaft inlet.

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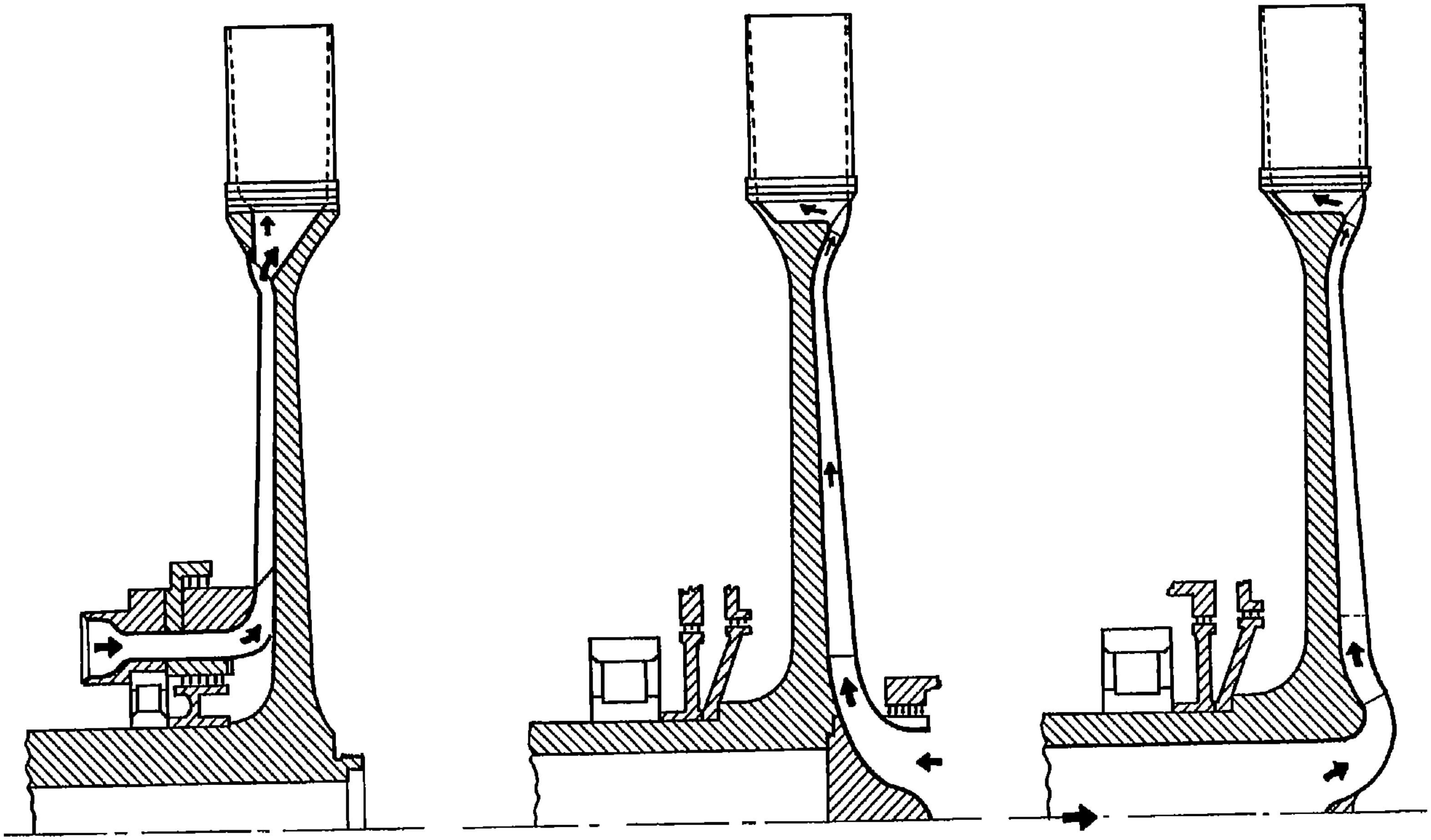
Figure 16. - Split-disk air-cooled turbine configurations.



(a) Upstream inlet. (b) Downstream inlet. (c) Hollow-shaft inlet.

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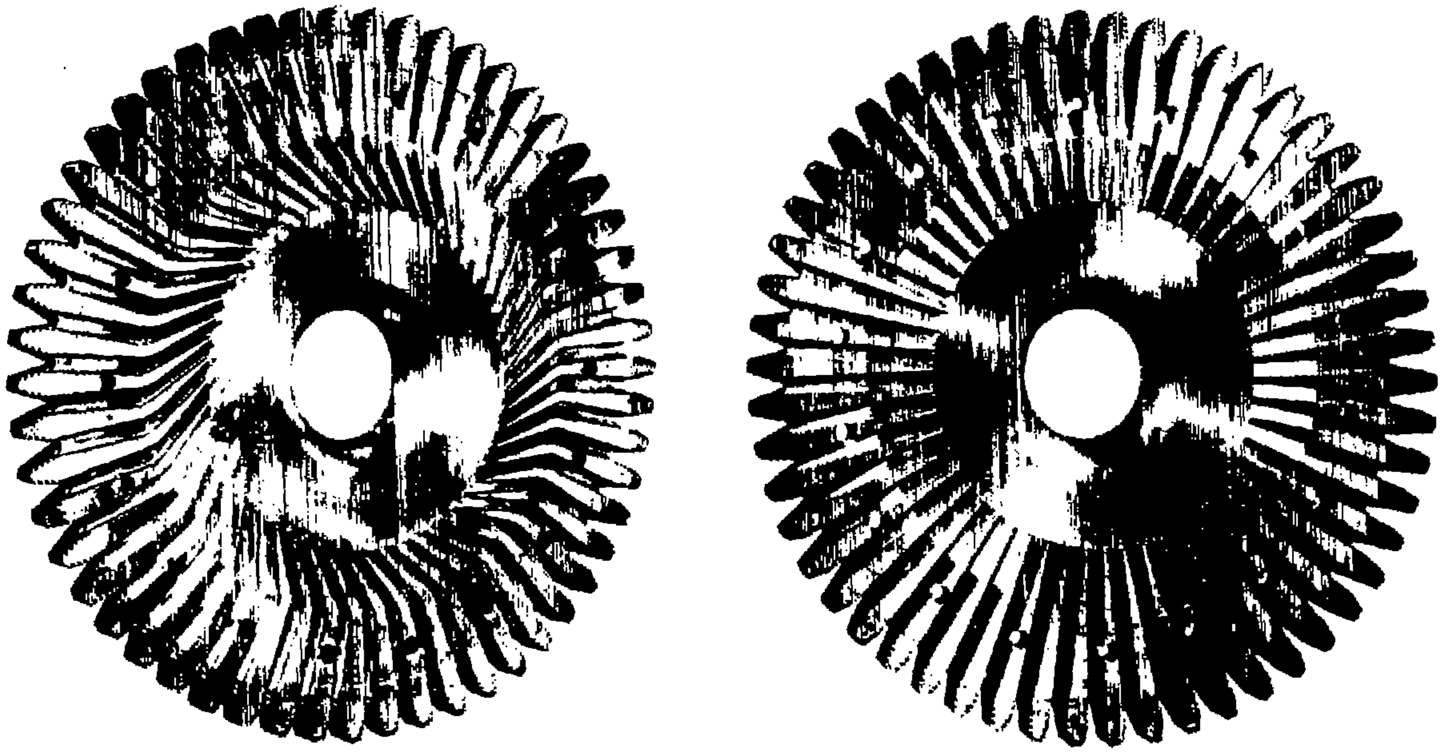
Figure 17. - Shrouded air-cooled turbine disk configurations.



Curved-vane inducer section

Straight vanes

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Figure 18. - Turbine cooling-air vane configurations.

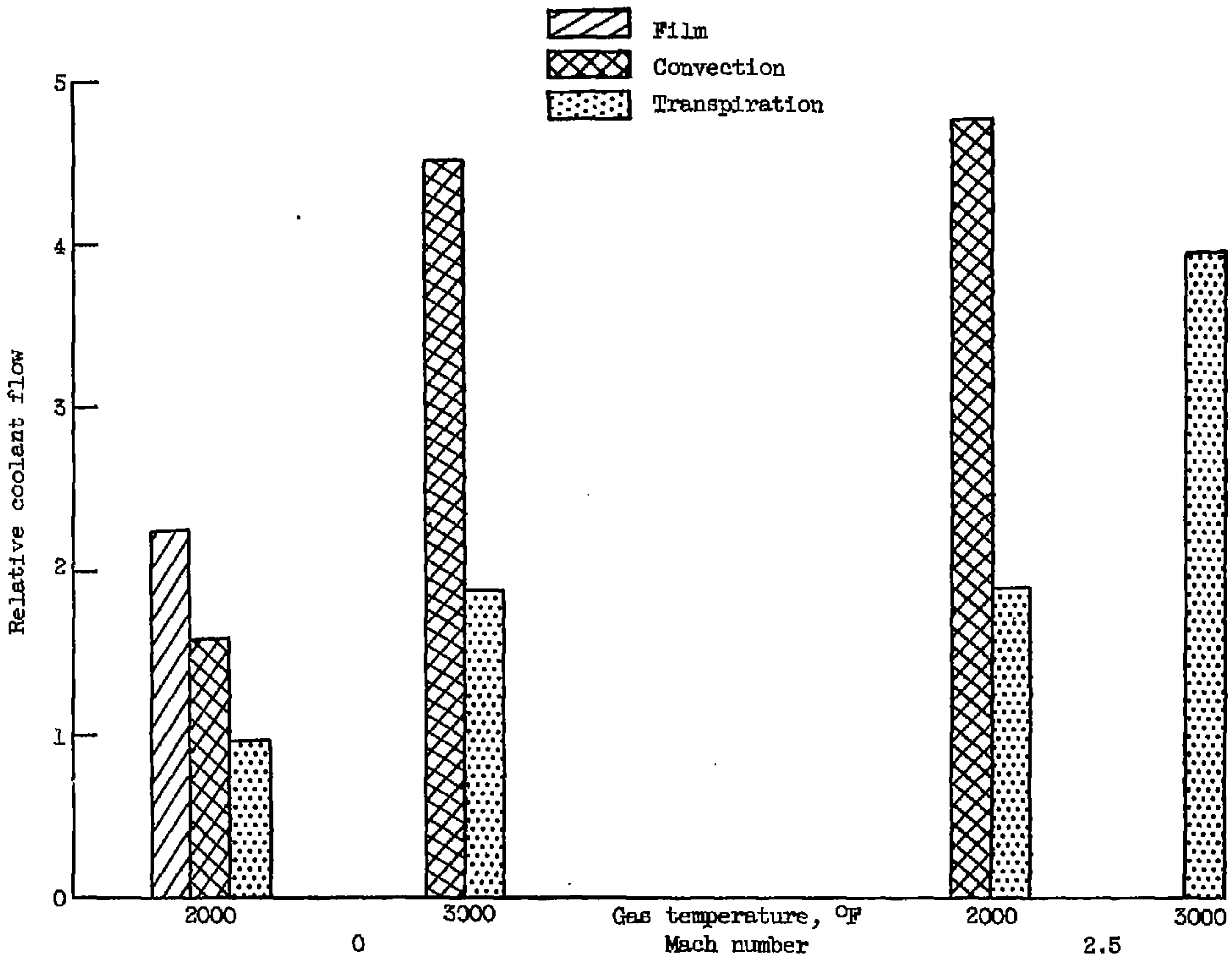

Figure 19. - Comparison of cooling effectiveness of three methods of air-cooling for two turbine-inlet temperatures and flight Mach numbers. Cooling air bled from compressor discharge.

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RESEARCH MEMORANDUM

REVIEW OF STATUS, METHODS, AND POTENTIALS OF GAS-TURBINE AIR-COOLING

By Jack B. Esgar and Robert R. Ziemer

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

February 23, 1955

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SUMMARY

The use of turbine cooling to allow increased turbine-inlet temperatures in gas-turbine engines permits greatly increased engine power output (thrust or shaft horsepower) and, for some types of engines, permits improved specific fuel consumption. In addition, cooling allows a greater degree of freedom in turbine design because of higher permissible stress levels and a greater range of possible turbine materials. The attainment of these benefits from turbine cooling is accompanied by a small performance reduction relative to the ideal uncooled engine performance as a result of cooling losses. For nonafterburning turbojet engines operating at a flight Mach number of 2 and constant turbine-inlet temperature, approximately a 1-percent reduction in thrust accompanies each percent of air bled from the compressor exit for turbine cooling. The effect on specific fuel consumption is generally negligible. The prevalent practice of bleeding air overboard for cabin cooling, driving accessories, and so forth, results in a thrust reduction more than double that caused by turbine cooling and causes substantial increases in specific fuel consumption. The power reduction resulting from air-cooling a turboprop engine at subsonic speeds and constant turbine-inlet temperature is somewhat higher than for a turbojet engine, but the net gains in power resulting from higher turbine-inlet temperatures are still very large. In addition, operation at the higher turbine-inlet temperatures can result in an actual decrease in specific fuel consumption, including the effects of cooling, relative to low-temperature uncooled engines.

At present, the most promising air-cooled turbine blades for conventional turbojet engines are convection-cooled corrugated-insert and strut-supported blades. Successful analytical methods of predicting blade temperatures have been found for both blades. General heat-transfer analyses indicate that these blades will probably be satisfactory except for operation under the combination of very high gas temperatures (2500° F) and high flight Mach numbers (above 2), where transpiration-cooling may be required. For small or thin blades, the maximum permissible temperatures and flight Mach numbers may be lower.

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INTRODUCTION

Past history has shown that, in order to obtain desirable cycle temperatures for heat engines, it was often necessary to cool certain engine components to circumvent material strength limitations. Thermodynamically, removal of heat from a cycle by cooling is detrimental to performance; but, practically, cooling permits a type of engine operation resulting in performance unattainable without cooling. An excellent example is the piston engine, where refinements in cooling methods, involving the removal of large quantities of heat, led to increasingly superior performance. It can be expected that a similar type of evolution will occur in the cooling of gas-turbine engines and result in a type of operation presenting many new performance possibilities.

There are three main purposes for cooling the turbines of gas-turbine engines. The first and most commonly accepted purpose is that higher engine cycle temperatures (resulting in increased specific power) can be obtained if means are provided for controlling the turbine disk and blade temperatures independently of the turbine-inlet gas temperature. The second purpose is to permit the use of higher operating stresses or to give longer turbine life by reducing the operating metal temperature of the turbine. Higher operating stresses allow a much greater amount of freedom in the turbine design and generally result in increased power for a given engine frontal area, because the flow capacity of the turbine can be increased through the use of longer turbine blades. The third purpose of turbine cooling is to permit a greater degree of freedom in the choice of turbine materials than is presently available to engine designers. Materials currently used for gas turbines are chosen for strength characteristics at high operating temperatures. In general, these materials also contain relatively large quantities of scarce or critical alloying elements. Lowering the turbine blade temperature permits the use of other materials that, in addition to having the capacity of withstanding higher stress levels, have reduced amounts of critical alloying elements relative to the so-called "high-temperature alloys."

This report discusses the use of less critical turbine materials and presents some of the performance possibilities attainable when turbine cooling is employed to permit use of higher cycle temperatures and higher turbine stresses. In addition, some of the progress that has been made in turbine air-cooling research is described, and an indication is given as to the expected trends in the development of cooled gas-turbine engines.

POTENTIAL ENGINE PERFORMANCE AT HIGHER TURBINE-INLET TEMPERATURES

Turbojet Engines

The thrust of a turbojet engine is essentially proportional to the product of the weight flow of air that passes through the engine and the velocity of the jet at the exhaust nozzle. All methods of increasing the thrust output, therefore, depend upon increasing either one or both of these variables in some manner. The weight flow of air per unit frontal area can be increased through the use of recently developed compressors. The jet velocity can be increased through the use of (1) higher compressor pressure ratios, higher turbine efficiencies, or lower burner pressure losses, all of which result in a higher available pressure in the exhaust nozzle and consequently allow higher expansion ratios for obtaining higher velocities, (2) improved exhaust-nozzle efficiencies, and (3) increased jet-exhaust temperatures.

Several of these effects on engine performance are illustrated in figure 1 (data from ref. 1) for both nonafterburning and afterburning turbojet engines operating at a flight Mach number of 2 in the stratosphere. With the newer compressors, the flow per unit of engine frontal area is often determined by the turbine. For this reason, the relative engine thrust is given per unit of turbine frontal area in figure 1. For this case, high compressor pressure ratios result in higher gas densities at the turbine and consequently increased flow capacity. (The compressor pressure ratios shown are for the actual operating conditions and not sea-level static.)

It can be seen from the figure that, for nonafterburning engines, turbine-inlet temperature has a very significant effect on thrust. Increases in turbine-inlet temperature can increase the thrust by a factor of 2 or more relative to that of present engines. At constant compressor pressure ratio, increases in turbine-inlet temperature generally result in increased specific fuel consumption. These increases can be explained by the fact that the thrust is increased directly as the jet velocity is increased, but the kinetic energy of the gases is increased as the square of the jet velocity. The fuel consumption is proportional to the kinetic-energy increase; therefore, the fuel consumption increases at a greater rate than the thrust, with a resultant increase in thrust specific fuel consumption. This effect can be largely compensated for by increasing the engine thermal efficiency through the use of higher compressor pressure ratios.

Figure 1 also shows that very high thrust levels are obtainable for afterburning engines. This thrust is obtained at a relatively high cost in specific fuel consumption, particularly at the turbine-inlet temperatures of current engines. If the turbine-inlet temperature is increased, a smaller pressure drop is incurred across the turbine, less fuel is burned in the exhaust nozzle (although more is burned ahead of the turbine), and the thrust output of the engine is increased at a substantial saving in specific fuel consumption. The advantage of increased turbine-inlet temperatures for afterburning engines is therefore primarily to decrease the fuel consumption. In addition, higher compressor pressure ratios combined with high turbine-inlet temperatures give very large increases in thrust and improved specific fuel consumption.

Turboprop Engines 3511

The power output of the turboprop engine, or other shaft-power turbine engines, is primarily a function of energy level of the gases ahead of the turbine; therefore, the turbine-inlet temperature has a direct bearing on the power. In the turboprop engine the gases are expanded almost completely in order to extract the maximum power, so that the pressure level and density at the last stage of the turbine are always low. Therefore, turbine flow capacity (weight flow per unit turbine frontal area) is almost independent of compressor pressure ratio. Because of this fact, power per unit turbine frontal area has little significance; of greatest interest is power per pound of compressor air, defined as specific horsepower.

The effects of turbine-inlet temperature and compressor pressure ratio on relative specific horsepower and specific fuel consumption are shown in figure 2. Here, as in figure 1 for the turbojet engine, the power output can be increased by a factor of 2 or more relative to present engines by increasing the turbine-inlet temperature. It will be noted, however, that opposite to the case for the nonafterburning turbojet, increasing turbine-inlet temperature decreases specific fuel consumption for the turboprop. A rigorous explanation of this trend is somewhat involved, but basically the reason for the decrease in specific fuel consumption with increasing turbine-inlet temperature is that both the gross turbine power and the fuel-flow rate are directly proportional to the turbine-inlet temperature. The net turbine power, or shaft power, is the gross turbine power minus the compressor power, and therefore it increases with turbine-inlet temperature at a rate proportionally greater than for the gross power. As a result, the specific fuel consumption, which is the ratio of fuel-flow rate to shaft power, decreases with increasing turbine-inlet temperature.

DESIGN FREEDOM OBTAINABLE WITH TURBINE COOLING

The flow capacity of modern compressors is rapidly increasing to the point where components other than the compressor (inlet diffuser, primary burner, turbine, afterburner, or exhaust nozzle) will determine the engine frontal area. For some applications, it appears that the turbine may have the largest diameter of any component of the engine. The flow area of the turbine can be increased by the use of longer turbine blades, but this often results in stresses in excess of those permissible with presently available materials in uncooled turbines. An indication of the manner in which turbine diameter is related to turbine blade stress for a given compressor weight flow and pressure ratio is shown in figure 3. Increasing the blade root stress from 30,000 to 60,000 pounds per square inch can result in a reduction in single-stage turbine diameter of approximately 15 percent. This corresponds to a frontal-area reduction of over 25 percent.

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Properties of some materials that can be used to obtain higher turbine stress levels are indicated in figure 4. For the temperatures at which gas-turbine blades operate, stress-rupture is the criterion that usually determines allowable blade stress. The stress-rupture properties of several materials are shown, and the upper levels of the curves are cut off where stress-rupture properties no longer determine the permissible stress. The alloy S-816 is commonly used in present gas-turbine engines. At a temperature of 1500° F (about standard blade temperature for present engines), the maximum allowable stress is 24,000 pounds per square inch. If, however, the temperature is reduced only 100° F by cooling, the allowable stress can be increased by about 35 percent, with further increases obtainable at lower temperatures. Below temperatures of about 1200° F, however, other materials, such as A-286, possess better strength properties, with the possibility of operating at stresses over 90,000 pounds per square inch - over $3\frac{1}{2}$ times the allowable stress for present engines.

Further reduction in temperature makes possible the utilization of high-strength steels such as Timken 17-22A(S). This type of material offers only slight increases in possible operating stress over A-286, but the critical-material content of 17-22A(S) is almost completely eliminated, the alloy containing about 97 percent iron. Even A-286 is presently considered a relatively noncritical alloy, because it is over 50 percent iron and contains no cobalt or columbium. Currently used blade materials such as S-816 contain very high quantities of critical materials such as cobalt, nickel, chromium, and columbium, the iron content of S-816 being only 2.8 percent.

In general, materials that exhibit high strength at temperatures from 1000° to 1300° F contain considerably smaller quantities of critical materials than the currently used high-temperature alloys. The significance of this fact is that cooling, in addition to permitting the use of higher turbine-inlet temperatures and stresses in turbine engines, permits the production of a greater number of engines because of the greater availability of suitable turbine materials. The factor of material availability was the primary reason the Germans used turbine cooling in some of their turbojet engines in World War II.

The very high stresses apparently available through the use of turbine cooling (fig. 4) do not necessarily mean that it is possible to operate the turbine blades at these stress levels. As discussed in references 2 and 3, a factor of safety, defined as a stress-ratio factor, is required in the design of air-cooled turbine blades. This factor of safety differs little from that used in standard design practice, except that the actual stress-rupture properties of the materials in air-cooled turbine blades are not known after blade fabrication. The stress-rupture properties shown in figure 4 are for bar stock. In air-cooled turbine blades, the metal is thin and the structures are usually brazed; both the thin metal wall and the brazing have the effect of decreasing the stress-rupture properties. This effect is believed to vary greatly with materials and brazing alloys and techniques used. Reference 3 showed that, for one type of blade design made of alloy 17-22A(S), the required stress-ratio factor was 2.3. This is the ratio of the stress-rupture for bar stock to the centrifugal stress in the failure area of the air-cooled turbine blade. It is believed that the use of other blade materials and possibly of other fabrication techniques may result in much lower values of stress-ratio factor. Even with high values of stress-ratio factor, higher turbine operating stresses are permissible through the use of turbine cooling than are presently possible with uncooled turbine blades.

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EFFECT OF TURBINE COOLING ON ENGINE PERFORMANCE

The most probable source of air for turbine cooling is the engine compressor. It is often possible to bleed the compressor air from some intermediate stage in order to keep the cooling-air temperature and the compressor work on the cooling air at a minimum, but for turbine rotor cooling it is more often necessary to bleed from the discharge of the compressor in order to obtain the highest possible pressure for effective blade cooling. The use of this air for cooling affects engine performance in several ways. The air is removed from part of the engine cycle so that it is unavailable for doing work in the turbine, but the work done on the cooling air makes the required specific turbine work higher than without bleed. When the air is used for turbine rotor cooling, additional work is done on the cooling air to accelerate it to the wheel speed at the blade tip as it passes through the rotor. For a given value of turbine-inlet temperature, therefore, the pressure and temperature ratios across the turbine will be somewhat higher when air is bled from the compressor.

Reduction of the exhaust-gas temperature due to dilution by the cooling air results in a thrust reduction compared with the case with no cooling air, but the mass-flow addition in the exhaust jet due to the cooling air results in higher thrusts than if the air were not mixed into the exhaust gases.

Turbojet-Engine Performance

Present knowledge of the quantities of cooling air required for operation at higher turbine-inlet temperatures may be used to predict the engine performance attainable with air-cooled turbines. Turbojet-engine performance with no consideration given to the effects of bleeding part of the compressor air for turbine cooling was discussed in connection with figure 1. The combined effect of increasing the turbine-inlet temperature and bleeding air from the discharge of the compressor to cool the turbine is shown in figure 5. These analytical results are based on the coolant flows that were calculated to be required for single-stage turbines with corrugated-insert blades. The total coolant flows for both turbine stator and rotor were approximately 3 percent at 2000° F, and 8 percent at 2500° F. These are probably about the minimum flows that could be expected with corrugated-insert blades, but should be easily within the capabilities of other blades to be discussed later. Cooling generally results in a small decrease in thrust with practically no sacrifice in fuel consumption relative to the calculated performance without cooling for nonafterburning engines. In afterburning engines, the effect of cooling is to increase fuel consumption with only small effects on thrust. These effects can be explained by the fact that cooling generally shifts the performance map in the direction of lower turbine-inlet temperatures by diluting the exhaust gases and lowering the temperature downstream of the turbine.

Turboprop-Engine Performance

<u>Air-cooling</u>. - The predicted performance of an air-cooled turboprop engine at sea-level static conditions is shown in figure 6. The cooling-air requirements were assumed to be the same as for the turbojet engine. The decrease in performance relative to the calculated performance without cooling (fig. 2) is greater for the turboprop than for the turbojet engine, because in the turboprop the cooling air passing through a cooled turbine stage is completely lost for engine power generation in that stage of the turbine. In the turbojet engine, on the other hand, the cooling air is still available for obtaining power in the form of jet thrust. At turbine-inlet temperatures up to 2000° F with the turboprop engine, however, the specific fuel consumption decreases with increasing values of turbine-inlet temperature even when the effects of cooling are included.

<u>Liquid-cooling</u>. - Also shown in figure 6 is a performance point for a liquid-cooled turbine under the most severe conditions given. In this case the only effect of cooling on the cycle is the removal of a small portion of heat from the gases. The effect on performance is extremely small; therefore, liquid-cooling of turboprop engines is very promising if radiator drag losses are tolerable. Whether the radiators will cause

an acutal drag is open to question and will be determined by the radiator design. The airplane design will, of course, be materially affected by requirements for ducting air aboard for the radiator. For stationary power plants, the radiator usually does not present a problem; consequently, liquid-cooling is extremely promising for this use. Even though the performance of air-cooled turboprop engines is inferior to the performance of liquid-cooled engines, the use of air-cooling is very promising for aircraft, because substantial increases in power are attainable at no increase in specific fuel consumption relative to uncooled engines at current gas-temperature levels, and no radiators are required. 3511

Performance Variations with Compressor-Discharge Air Bleed

Previous figures have shown the performance potentialities attainable through the use of turbine cooling to permit engine operation at higher turbine-inlet temperatures. It is also important to study the engine performance variations resulting from turbine cooling at a constant turbine-inlet temperature for various quantities of cooling air to determine the effort that must be expended in finding methods that will require smaller quantities of coolant and that will reduce losses.

The effect of bleeding various amounts of air from the compressor for cooling or other purposes is shown in figures 7 and 8 for nonafterburning turbojet and turboprop engines, respectively. The quantity of air required for cooling depends upon the type of air-cooled blades used in the engine; therefore, at given engine conditions a wide variation in cooling-air requirements is possible. In most aircraft gas-turbine engines, air is bled from the compressor for such uses as accessory drives or cabin cooling, in which cases the air is thrown overboard and cannot be used for jet thrust or turbine power. As a basis of comparison, the effects on performance of this prevalent practice of overboard bleed are shown in addition to the effects of bleeding air for turbine-cooling purposes.

The variations of relative specific thrust and specific fuel consumption with various percentages of cooling air bled from the compressor discharge for a turbojet engine operating at a flight Mach number of 2 in the stratosphere are shown in figure 7. The same general trends are obtained at conditions other than those given for this figure, except that, at lower turbine-inlet temperatures, losses due to cooling are somewhat higher. The thrust decreases approximately 1 percent for every percent of air bled from the compressor for turbine-cooling purposes. This loss is more than doubled if the air is bled overboard and cannot be used for jet thrust. Air bled for turbine cooling has only a very slight effect on specific fuel consumption, but overboard bleed increases the specific fuel consumption over 2 percent for every percent of bleed air.

The effect of bleeding various amounts of cooling air on the performance of a turboprop or a stationary power plant operating at sea-level static conditions (no jet thrust horsepower) is shown in figure 8. As explained in connection with figure 6, the decrease in performance due to air-cooling at a constant turbine-inlet temperature is larger for a turboprop than for a turbojet engine. This is particularly true concerning specific fuel consumption, for which the increase with cooling-air flow is more than 5 times as high as for a turbojet engine. The decrease in power due to bleeding air for turbine cooling is only slightly higher than the decrease in thrust for a turbojet. As stated previously, however, operation at the higher turbine-inlet temperatures permitted by turbine cooling can result in an actual decrease in specific fuel consumption, including the effects of cooling, relative to low-temperature uncooled engines. For turboprop engines, bleeding air for turbine cooling is only slightly less costly than for cabin cooling, accessory drives, electronic equipment cooling, and so forth.

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As previously discussed, the effect of liquid-cooling on turboprop or stationary power-plant performance is extremely small. The change in power or specific fuel consumption caused by liquid-cooling at the conditions of figure 6 is less than 0.5 percent. For aircraft application, of course, the effect of radiator drag on over-all performance would also have to be included.

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As explained in reference 4, there is an optimum jet velocity for turboprop engines at each flight speed that results in minimum fuel consumption and maximum thrust. The performance variations due to bleeding air from the compressor for cooling purposes are somewhat smaller for turboprop engines with optimum jet thrust than for the case with no jet thrust, but the improvement is of little significance.

Up to the present time it has been impossible to verify experimentally all the effects of cooling on engine performance. Probably the largest unknown in the prediction of engine performance is the effect on turbine aerodynamic performance of discharging cooling air from the blade tips. Tests at the NACA, while not conclusive, do not show a measurable effect of the cooling-air discharge on turbine efficiency. Tests conducted by the British (ref. 5) show that a coolant flow of 2 percent of the compressor flow to each the turbine rotor and the turbine stator affected the turbine stage efficiency less than 0.5 percent. In the British investigation, the stator cooling air was discharged at the stator inner diameter ahead of the turbine rotor. If, instead, this air had been ducted to mix with the exhaust gases downstream of the turbine, an even smaller effect on turbine efficiency could be expected.

The effect of discharging the rotor cooling air through several blade tip configurations was investigated in reference 6, which indicated that with the proper tip design a substantial portion of the pumping work on the cooling air in the turbine rotor can be recovered in the form of added turbine work. Generally, the effect of this turbine work on the over-all performance is of rather small significance, as illustrated in figures 7 and 8. The difference between the stator and rotor cooling curves is due to the pumping work in the turbine rotor. The experimental studies just mentioned indicate that the methods used for predicting cooled-engine performance are reasonably accurate and the trends shown should be correct.

RESEARCH ON AIR-COOLED TURBINE BLADES

Both air- and liquid-cooling methods have their relative advantages and disadvantages. Most NACA research on liquid-cooling has been of a fundamental nature to determine the laws governing heat transfer within the cooling passages. This research is summarized in reference 7, and more recent research on natural-convection cooling is presented in reference 8. Liquid-cooling research on actual turbine blade configurations and systems has been very limited. Since air-cooling appears to be a quite promising method for turbojet engines and more research information is available than for liquid-cooling, the discussion of methods of cooling is limited to that type.

Air-Cooling Methods

When air-cooling is incorporated in an engine, the entire engine design must be considered so that the air will be utilized most effectively. A possible engine configuration is shown in figure 9. Air is ducted from the discharge or one of the rear stages of the compressor to cool the turbine rotor and stator and is then discharged into the gas stream downstream of the turbine to provide additional thrust. A hollow turbine shaft provides a very convenient method of ducting cooling air to the rotor and can often eliminate many air-sealing problems.

Three methods of air-cooling are shown in figure 10. The most conventional method used in all heat-transfer processes is convection cooling (fig. 10(a)). With this method, it is desirable to increase heat-transfer surface area on the heat-rejection side of the apparatus such as in the form of the fins shown. This method of cooling has been successfully used on air-cooled piston engines for many years. The method of cooling shown in figure 10(b) is less well known. A film of cool air is introduced through slots to form an insulating layer between the hot gases and the cooled surface. The thermal conductivity of air is very low, so that it is a good insulation medium, but the effectiveness of the layer of air is lost some distance downstream of the slot when it mixes with the hot gases. This disadvantage is eliminated by transpiration cooling (fig. 10(c)), because air is continuously bled through the entire area of a porous surface. Transpiration cooling is the most effective method of air-cooling known at the present time. A comparison of the cooling effectiveness of these three methods of cooling is given in reference 9.

Air-cooled turbine blades of about 2-inch chord utilizing these various methods of cooling are illustrated in figure 11. The hollow blade was used by the Germans in some of their engines in 1945. A survey of their work on cooling turbojet engines and turbosuperchargers is given in references 10 to 12. The cooling effectiveness of the hollow blade is so low that excessive quantities of cooling air are required; consequently, efforts were made to provide added internal heat-transfer surface area. The tube-filled blade was an early attempt of the NACA to provide this extra surface area. Some results of tests on and methods of manufacturing this type of blade are given in references 2, 3, and 13 to 17. Although more recent blade developments have led to superior blade configurations so that the tube-filled blade configuration is presently of little interest, much valuable information has been obtained from this configuration: (a) The feasibility of building blades of nonstrategic materials was demonstrated (refs. 2, 3, 15, and 16); (b) suitable coatings for nonstrategic materials were investigated (ref. 17); (c) methods of blade fabrication including forming, brazing, heat-treatment, and suitable types of structure were studied (refs. 2 and 15); and (d) a relation was obtained between bar-stock stress-rupture life and blade life for nonstrategic blades made of 17-22A(S) steel (ref. 3).

The British (ref. 18) have used a somewhat different method of approach to the problem of adding internal surface area. Instead of packing a hollow shell with tubes and brazing the assembly together, they drilled holes in a solid blade to provide coolant passages with a greater internal surface area than is possible in a hollow blade.

Cooling of the leading and trailing edges is often difficult with tube-filled blades. In an attempt to improve cooling effectiveness in these regions, film cooling was investigated on the type of blade shown in figure 11(c), with the results reported in references 19 to 21. Cooling of the leading and trailing edges was effective, but blade durability was a serious problem (ref. 16). Research was conducted in Germany on blades having film cooling around the complete periphery (ref. 22), and similar blades have been built in this country. Although cooling is adequate for some cases, durability is usually poor. Another solution to the problem of leading- and trailing-edge cooling is to increase the thermal conductivity of the blade shell; consequently, shells that were copper-clad on the inside surface (fig. 11(d)) were investigated in reference 20. This type of structure is similar to that of copper-clad kitchen utensils in which the copper spreads the heat over the entire area of the utensil. The biggest disadvantage of the copper-clad blade is the weight increase, which raises the stress so much that the gains from cooling are practically eliminated.

A practical type of shell-supported convection-cooled blade construction is the corrugated-insert blade (fig. 11(e)). Large amounts of heat-transfer surface area can be added in the form of corrugated fins, and the fins can be made to extend well into the leading and trailing edges of the blade to ensure adequate cooling in these regions. An island is usually provided in the middle of the passage so that the corrugations can be of uniform amplitude. This island is blocked off from the cooling air. The island can be eliminated in small blades, but the corrugations will not be of uniform amplitude. Temperature data obtained from a corrugated blade are discussed in the next section.

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In all the turbine blades discussed up to this point, the blade shell has been the primary support member for carrying the stresses due to centrifugal forces. Since the shell is exposed to the gas stream, it is also the hottest member of the blade; and, therefore, its stress-carrying capacity is lower than that of cooler portions of the blade. For this reason, blades have been designed and tested with the main stress-carrying member, or strut, submerged inside the coolant passage and operating at a lower temperature than the blade shell (figs. 11(f) and 12). The shell can be made thin and can be completely supported by the strut. In this manner the stresses in the shell are greatly reduced, and it can operate at higher temperatures. With higher shell temperatures, the heat transfer from the gas to the blade is reduced and the quantity of cooling air required is reduced. This type of blade shows great promise for future application in air-cooled turbine engines.

The strut-supported blade shown in figure 12 and investigated in reference 23 is an example of only one of several possible methods of construction. The blade shown can be fabricated by machining the component parts of the strut and base and then brazing the final assembly together. The shell can be attached by brazing or spot-welding. Research is currently being conducted on a strut-blade configuration having the strut cast in an integral piece. The final configuration is approximately the same as the one shown in the assembled view in figure 12.

The blade in figure 11(g) is a transpiration-cooled blade. The porous shell could be made from several materials, the most probable being woven wire cloth or porous sintered materials made from powdered metal. Only a limited amount of experimental data is presently available for transpiration-cooled turbine blades. Some results are given in references 24 to 27. Advantages and problems in the use of transpiration cooling are discussed in reference 28. Reference 29 states that, if transpiration-cooled blades are to operate satisfactorily over wide ranges of flight altitude and flight Mach number, the cooling air must be metered to local positions on the blade surface. This can be accomplished by incorporating orifices in the turbine blade base as shown in figure 13. In addition to providing more uniform cooling over the wide range of operating conditions, this method of fabrication greatly simplifies blade fabrication, because metering the air at the orifices instead of through the porous surface permits fabrication of blades with uniform permeability around the blade periphery. The strut shown in figure 13 has the dual purpose of dividing the blade into compartments so that an orifice can meter air to local positions on the blade surface and providing a support member for the porous shell. Porous sintered shells usually require an internal support because of low shell strength, and woven wire shells need the support to provide rigidity.

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Experimental Temperature Data

Experimentally measured turbine blade temperatures are presented in figure 14 for the corrugated-insert blade. The coolant-flow ratio used as the abscissa is defined as the ratio of the air used for turbine-cooling purposes to the total flow of air through the compressor. For the uncooled condition the blade temperature is over 200° F lower than the turbine-inlet temperature, because the gas total temperature relative to the turbine rotor blades is less than the gas total temperature relative to the stator blades, as a result of high rotative speeds of the turbine and high gas velocities at the stator exit. The use of only 2 percent of the compressor air for turbine rotor cooling will reduce the blade temperature over 400° F below that of an uncooled blade, or approximately 650° F below the turbine-inlet temperature. These results show the substantial blade temperature reductions possible with very nominal amounts of cooling air. In addition, they show the success obtained in developing methods of predicting the average blade temperature of this type of blade and lend encouragement to using these methods for predicting cooling requirements for other conditions. The variation between measured and predicted temperatures is less than 35° F. The predictions are based essentially on the methods discussed in references 30 and 31.

A more complete summary of methods of analytically predicting air-cooled blade temperatures is presented in reference 7. In addition, a method is presented in reference 32 that provides a quick rough evaluation of the cooling effectiveness and pressure-drop characteristics of convection-cooled turbine blades. Methods are presented in reference 33 for rapidly evaluating the heat-transfer and pressure-drop characteristics of corrugated-insert air-cooled blades.

A comparison of the cooling effectiveness of a corrugated-insert blade with that of a strut-supported blade is given in figure 15. The temperature comparison is based on the temperatures of the stress-carrying members, which are the shell of the corrugated blade and the strut of the strut-supported blade. The data for the strut blade were obtained from reference 23. Figure 15 shows that the coolant flow required for the strut-supported blade is about half that required for the corrugated blade in order to obtain a specified blade temperature. At the very low flow rates, this difference will have very little effect on the over-all engine performance; but the results of figures 7 and 8 show that, as the cooling load becomes more severe, the savings in cooling air with the strut-supported blade could result in appreciable gains in engine performance. Experimental and calculated temperatures for the strut-supported blades are also compared in figure 15. Again the agreement is very good. The calculated temperatures were obtained by the methods described in reference 34. 3511

Caution must be observed in generalizing the results obtained from research on the blades shown in figures 11 to 15. All these blades had a chord of approximately 2 inches or more and would be suitable for many turbojet engines. For some high-compressor-pressure-ratio turbojet engines and for most turboprop engines, however, the blades are much smaller. Research is presently being conducted on these smaller blades, and also on blades that have such thin cross sections that use of internal heat-transfer surface area is difficult. Preliminary studies indicate that scaling down some of the cooling schemes such as the strut blade appears to be feasible, but in addition new types of design will be required for many applications. The research has not advanced far enough to warrant discussion.

Air-cooled blade durability is equally as important as the blade temperature reduction that is possible by cooling. Some of the endurance investigations, conducted mostly on nonstrategic tube-filled blades having a critical blade section stress of about 23,500 pounds per square inch, are reported in references 2, 3, 16, and 17. Air-cooled turbine blade life of 350 hours at full-power engine conditions has been obtained with no indication of failure (ref. 3). Although further endurance programs are required on other types of blades, other blade materials, and blades operating at higher stress levels to obtain conclusive results, it is indicated that, with proper design, air-cooled turbine blade life should be satisfactory. Turbine blade stress levels in excess of 40,000 pounds per square inch are being investigated, and it is indicated that blade stresses as high as 50,000 pounds per square inch may be feasible.

Blade Pressure Losses

In addition to the heat-transfer characteristics of turbine blades, the pressure-loss characteristics are important, because they determine to a large extent the flow capacity of the blades and the point at which cooling air should be bled from the compressor. There is a general relation between heat transfer and friction, so that blades with high cooling effectiveness also have relatively high pressure losses. A quick method of calculating the pressure changes through air-cooled turbine blades is presented in reference 35. Friction factors for two air-cooled turbine blade configurations were measured and reported in reference 36, in which air-cooled blade pressure losses were calculated within ±6 percent.

The general practice in the design of air-cooled turbine blades has been to design for a maximum cooling-air Mach number of 0.5 at the blade tip. At a given inlet supply pressure, the flow at a Mach number of 0.5 is about 75 percent of the choked flow. It is felt that designing for higher Mach numbers leaves little margin of safety for extreme conditions that may be encountered in engine operation. A method of blade design that takes into account both pressure loss and heat transfer is presented in reference 33 for corrugated-insert blades.

AIR-COOLED TURBINE DISK CONFIGURATIONS

The use of air-cooled turbine blades will require a type of turbine rotor construction different from that in current use. There is, however, a considerable amount of freedom in the type of design possible. Two main types of turbine rotor are the split disk (fig. 16) and the shrouded disk (fig. 17). With either construction the cooling air can be supplied from the upstream direction, the downstream direction, or through a hollow turbine shaft. With any of these possible types of construction, internal vanes are required in the turbine rotor to direct and help pump the cooling air out to the blades. Vane configurations for two experimental split-disk air-cooled turbines are shown in figure 18. In one case the vanes were curved to provide an inducer section for the cooling air, while in the other case straight vanes were used. Experimental tests reported in reference 37 do not indicate the superiority of either type of vane construction, and in each case the pressure rise in the turbine disks is small. In general, the pressure supplied at the rotor hub will have to be as high as the static gas pressure at the turbine blade tips, or higher.

Up to the present time, experimental tests have been conducted on several turbines with the type of disk configuration shown in figure 16(b), and some of the results presented in references 38 to 40 indicate that disk cooling will be adequate with the amount of air required for blade cooling. Experimental results have not been obtained, however, at turbine-inlet temperatures in excess of 1900° F. In the experimental turbines the downstream inlet was required in order to minimize the alterations to a commercial engine when incorporating turbine cooling. A discussion of some of the relative merits of other types of disk construction is given in reference 41.

PROBABLE FUTURE USE OF VARIOUS TURBINE-COOLING METHODS

Most experimental research on turbine cooling has been concerned with investigating various possible types of blade configurations to determine cooling effectiveness, fabrication problems, and expected durability. Investigations have been conducted in commercial engines modified to accommodate air-cooled turbines, and the turbines were usually made of nonstrategic materials. A combination of this experimental research and analysis has made possible the verification of analytical procedures, and cycle calculations have been made to determine the areas of operation in which future use of turbine cooling will be most profitable. The use of cooling is particularly promising for turbojet engines powering aircraft at supersonic speeds and for turboprop engines or other shaft-power turbine engines used in subsonic and transonic flight and stationary or marine power plants.

The relative merits of three methods of blade cooling (convection, film, and transpiration) for future use in cooled turbojet engines are indicated in figure 19. The relative coolant flows required for these various types of cooling are shown for turbine-inlet temperatures of 2000° and 3000° F and for flight Mach numbers of zero and 2.5. The results can be used to indicate trends only, because they are not the result of a design study. No consideration is given to cooling-air pressures that may be required or to whether actual blades will be feasible for the various conditions based on size, durability, and so forth. The calculations are based on an assumed blade temperature of 1250° F, compressor-discharge air bleed with no refrigeration of the cooling air, and results presented in reference 9. It was assumed that the compressor had a pressure ratio of 10 at a Mach number of zero and a pressure ratio of 6 at a Mach number of 2.5. It should be remembered that no experimental information is available concerning cooling at a gas temperature as high as 3000° F; consequently, results at 3000° F are based on the extrapolation of correlation methods obtained at lower temperature levels. The results shown at this temperature, therefore, must be considered approximate.

At low flight speeds figure 19 shows that the quantity of coolant required for turbine-inlet temperatures up to 2000° F (almost 400° F above current practice) is relatively small for either convection or transpiration cooling. Film cooling appears to be impractical mainly because of the problem of blade durability; but, in addition, the flow requirements are higher than for other methods. The convection-cooling bar is representative of the better shell-supported blades (see fig. 11(e)). Strut-supported-blade cooling-air requirements would be intermediate between those shown for convection and transpiration cooling.

As turbine-inlet temperature is increased to 3000° F at low flight speeds, the cooling requirements become more severe, and film cooling appears to be completely out of the question. Convection cooling is possible, but relative to transpiration cooling the requirements are high.

As flight speeds are increased, the ram-air temperature increases considerably, so that the temperature of the compressor-discharge air used for turbine cooling rises rapidly. The high cooling-air temperature makes turbine cooling more of a problem at high flight speeds. As shown in figure 19, at a flight Mach number of 2.5 the blade cooling problems are just as severe for a turbine-inlet temperature of 2000° F as they are at low flight speeds for a turbine-inlet temperature of 3000° F. Increasing the turbine-inlet temperature to 3000° F at a flight Mach number of 2.5 and using compressor-discharge air create a cooling problem so severe that transpiration cooling is the only air-cooling method presently known that could operate efficiently. It is possible, however, to provide a certain amount of refrigeration to the turbine-cooling air. Several systems including refrigeration for utilizing compressor air for turbine cooling at high flight Mach numbers are presented in reference 42. With refrigeration systems, convection cooling at high flight speeds and high temperatures is also feasible. At flight Mach numbers up to somewhat over 2.0, cooling-air refrigeration is probably unnecessary.

It can be concluded from this study and other studies that, for gas temperatures up to about 2500° F and flight Mach numbers up to at least 2.0, convection-cooled blades of the corrugated-insert and strut-supported types may be adequate. At higher temperatures and higher flight speeds, transpiration cooling may be required. The cooling of small or thin blades appears more difficult, and maximum temperature and flight speeds may be lower than for the cooling methods discussed.

There is also the question concerning the relative merits of air- and liquid-cooling. Because of its heat-transfer properties, water is one of the best liquid coolants and will probably find use in almost any type of liquid-cooling system. In most cases the heat picked up by the water will be rejected to ram air. At Mach numbers in excess of 2.0 in the stratosphere, the ram-air temperature is higher than the boiling point of water at reasonable pressures in the radiator. With water-cooled systems, therefore, the flight Mach number will be limited to approximately 2, or else systems will have to be designed that are capable of operating at extremely high water pressures - preferably above the critical pressure of water (3206 psi). Such high pressures would suggest the use of a rotating heat exchanger to eliminate the necessity of transferring the high-pressure water through seals from rotating to stationary parts of the engine.

Another problem associated with water-cooling systems in military aircraft is the vulnerability to enemy attack. Thus, reliability is somewhat more questionable than for air-cooled systems. A more complete description of several methods for and problems involved in dissipating the heat in a liquid-cooled turbine is given in reference 43. Cooling losses are always less (neglecting effects of radiator drag) with water-cooling than with air-cooling, but for turbojet engines the difference in performance probably is not of very great importance. Therefore, it is expected that air-cooling will find more use in turbojet applications. For very small turbojet-engine turbine blades it is possible, however, that better cooling could be obtained with liquids than with air.

For turboprop applications, air-cooling losses are larger than for 3511
turbojets, but substantial performance improvements are possible with either air- or liquid-cooling to permit higher turbine-inlet temperatures. In addition, the flight Mach numbers for turboprop aircraft will probably always be subsonic or transonic, so that heat rejection with liquid-cooling systems will not be a serious problem. For aircraft application, the better performance of liquid-cooled systems will have to be weighed against lower vulnerability of the air-cooled systems. Either type of cooling system can probably be utilized satisfactorily. For stationary or marine power plants, however, there appear to be no particular advantages of air-cooling over liquid-cooling as a method to permit high-temperature operation; consequently, liquid-cooling will probably be more satisfactory, because it will result in lower fuel-consumption rates. For either air- or liquid-cooling, the potential gains in engine performance through the use of higher turbine-inlet temperatures for both shaft- and jet-power turbine engines are well worth the effort required to build turbine cooling into the engine design.

CONCLUDING REMARKS

This review of the status, methods, and potentials of gas-turbine cooling can be summarized as follows:

Analytical Studies

1. The power output of nonafterburning turbojet engines and of turboprop engines can be increased by a factor of 2 or more by using turbine cooling to permit increases in the turbine-inlet temperature.

2. For afterburning turbojet engines, the effect of turbine cooling to permit increased turbine-inlet temperatures is primarily to decrease fuel consumption. In addition, for higher compressor pressure ratios combined with high turbine-inlet temperatures, large increases in thrust are obtainable at slightly improved specific fuel consumption.

3. Appreciable increases in turbine stress level are indicated through use of turbine cooling to reduce turbine metal temperature. Increased turbine stress levels allow reduction of turbine frontal area. The reduced metal temperatures also widen the choice of possible turbine materials, many of which contain only small amounts of critical alloying elements.

4. For nonafterburning turbojet engines operating at a flight Mach number of 2 in the stratosphere and constant turbine-inlet temperature of 2000° F, approximately a 1-percent loss in thrust accompanies each percent of air bled from the compressor exit for turbine cooling. The effect of turbine cooling on specific fuel consumption is negligible at most operating conditions for nonafterburning engines. The prevalent practice of bleeding air overboard for cabin cooling, accessory drives, and so forth results in thrust losses more than double the loss for turbine cooling, and in addition the specific fuel consumption increases over 2 percent for every percent of overboard bleed air.

5. For turboprop engines at subsonic speeds, the power reduction resulting from air-cooling at a constant turbine-inlet temperature of 2000° F is somewhat higher than for a turbojet engine, but the net gains in power resulting from higher turbine-inlet temperatures are still very large. In addition, operation at the higher turbine-inlet temperature can result in an actual decrease in specific fuel consumption, including the effects of cooling, relative to low-temperature uncooled engines.

6. Liquid-cooling has an extremely small effect on engine performance (neglecting radiator drag losses). In most cases, liquid-cooling will cause less than 1-percent loss in performance for turboprop engines.

7. Based on general heat-transfer analyses, it appears that convection-cooled corrugated-insert and strut-supported turbine blades will probably be satisfactory for most turbojet applications, except for operation under the combination of very high gas temperatures (2500° F) and high flight Mach numbers (above 2), where transpiration cooling may be required. For small or thin blades the maximum permissible temperatures and flight Mach numbers may be lower.

Experimental Studies

1. At present, the most promising types of air-cooled turbine blades for conventional turbojet engines are the corrugated-insert and strut-supported blades. Further research is required, and is under way, on both small and thin turbine blades. Preliminary studies indicate that scaling down some of the cooling schemes, such as the strut blade, appear to be feasible, but in addition new types of design will be required for many applications.

2. Analytical blade temperature predictions have been found to give reasonable agreement with experimentally measured temperatures for corrugated-insert and strut-supported turbine blades.

3. Endurance investigations have been conducted on a limited number of air-cooled turbine blades. Although further endurance programs are required on other types of blades, other blade materials, and blades operating at higher stress levels, in order to obtain conclusive results, it is indicated that with proper design air-cooled turbine blade life should be satisfactory.

4. Many types of air-cooled turbine disk designs appear to be feasible. Experimental results from several turbine disks of one type of design indicate that disk cooling will be adequate with the amount of air required for blade cooling. Experimental results have not been obtained, however, at turbine-inlet temperatures in excess of 1900° F. 3511

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, December 2, 1954.

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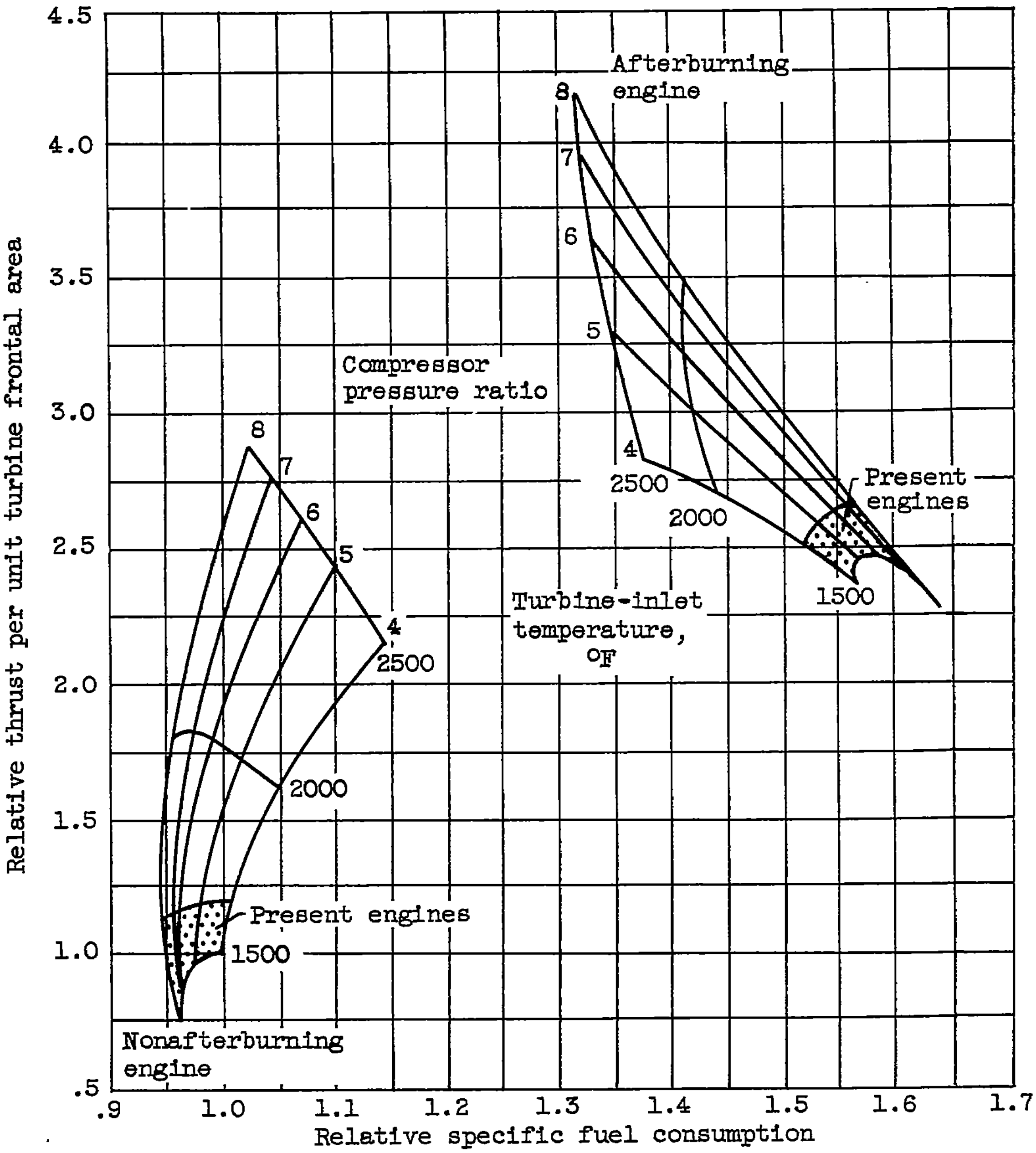

Figure 1. - Turbojet-engine performance for various turbine-inlet temperatures and compressor pressure ratios at flight Mach number of 2 in stratosphere (from ref. 1). Turbine hub-tip radius ratio, 0.75; afterburner gas temperature, 3000° F.

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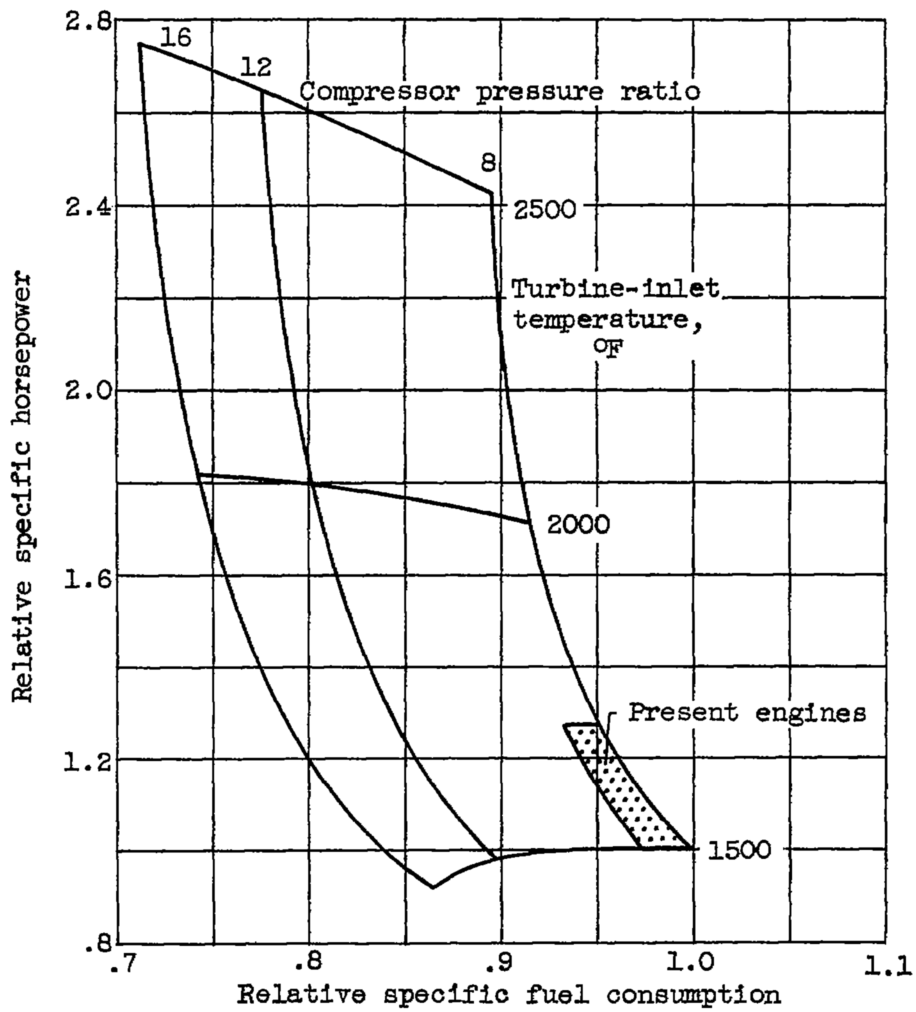

Figure 2. - Turboprop-engine performance for various turbine-inlet temperatures and compressor pressure ratios at sea-level static conditions.

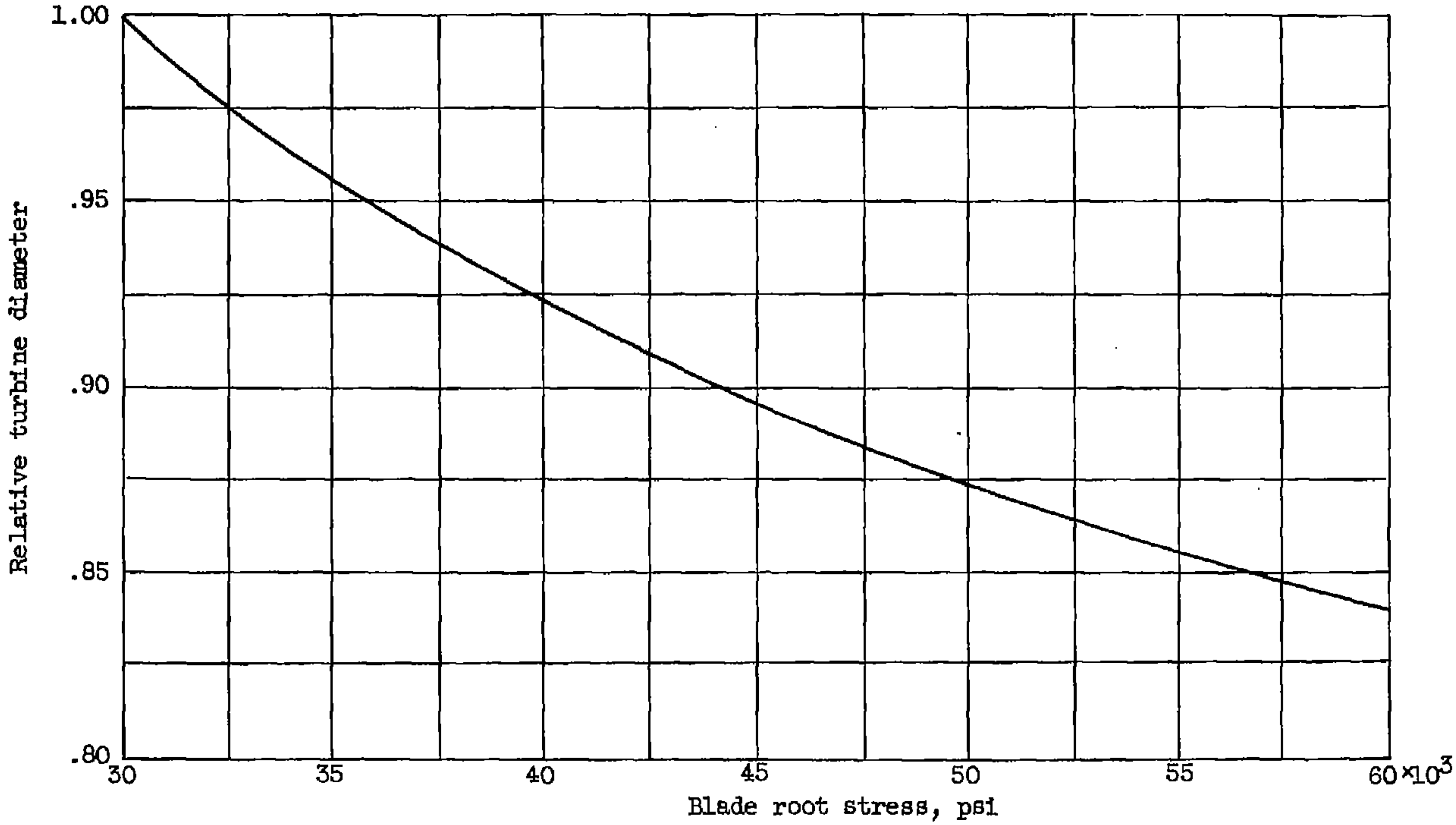

Figure 3. - Relation of turbine stresses to turbine diameter. High-output single-stage turbine for compressor pressure ratio of 7.

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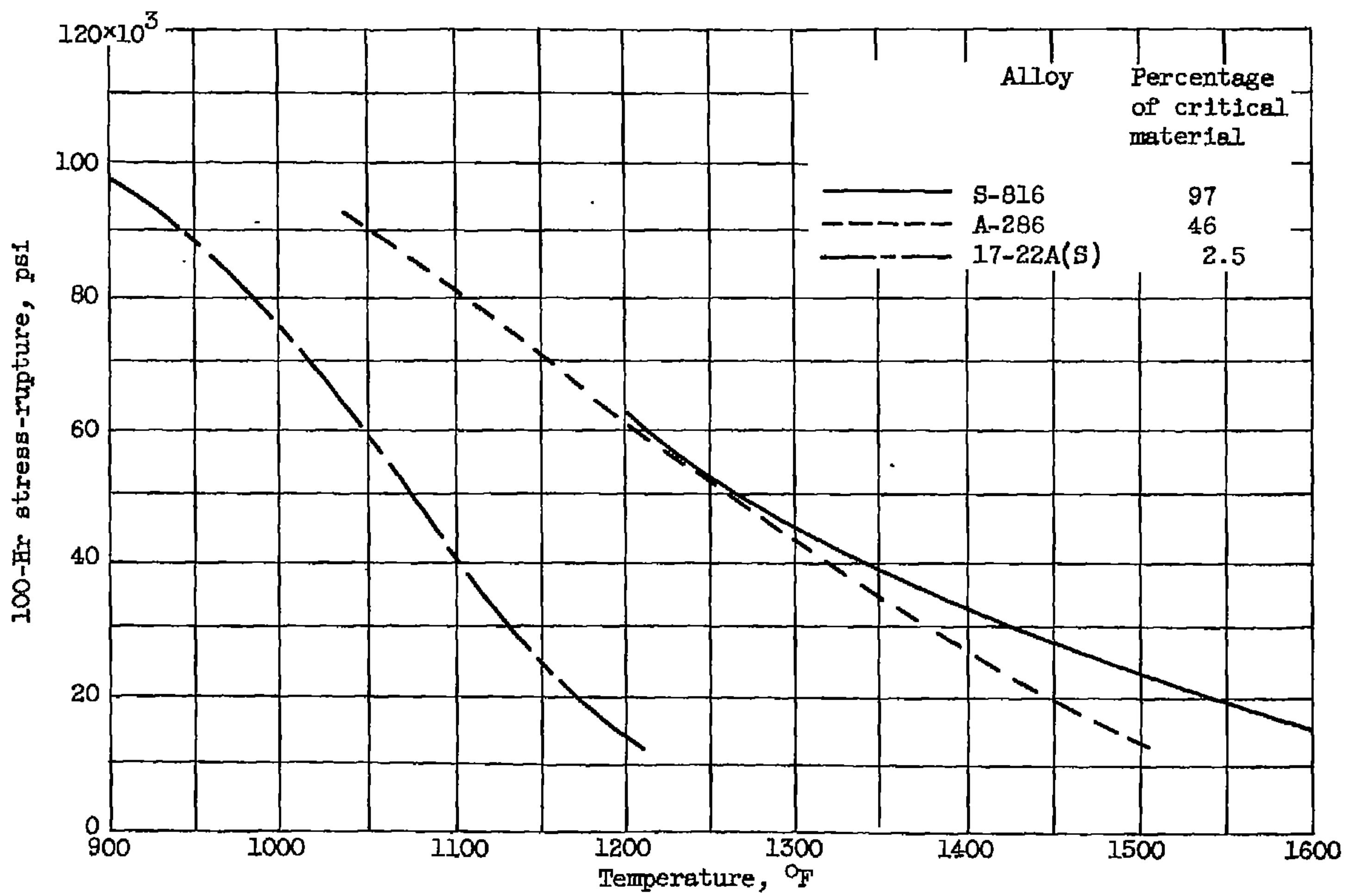

Figure 4. - Stress-rupture properties for possible air-cooled turbine blade materials.

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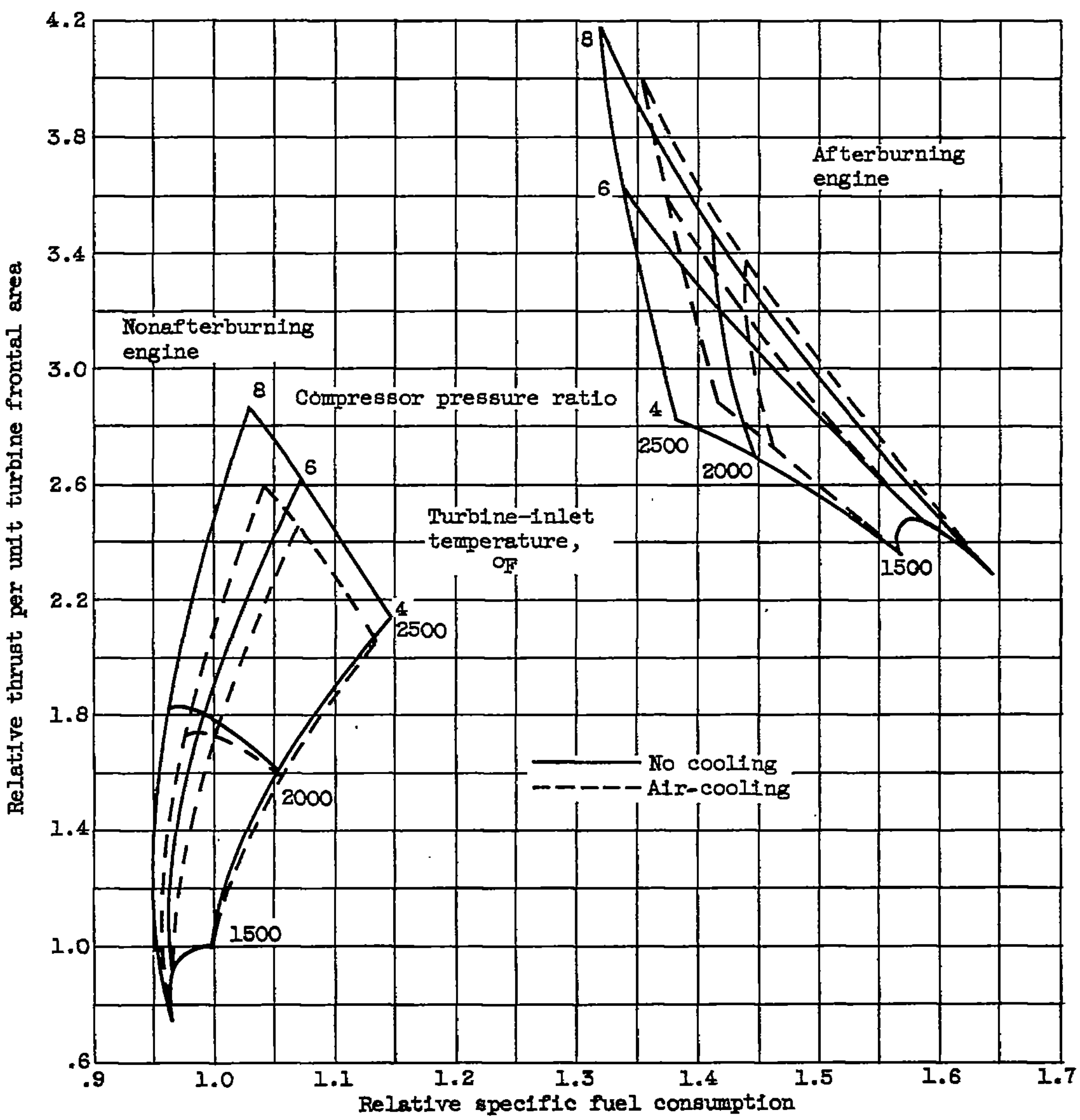

Figure 5. - Effect of air-cooling on turbojet-engine performance for various turbine-inlet temperatures and compressor pressure ratios at flight Mach number of 2 in stratosphere. Turbine hub-tip radius ratio, 0.75; afterburner gas temperature, 3000° F.

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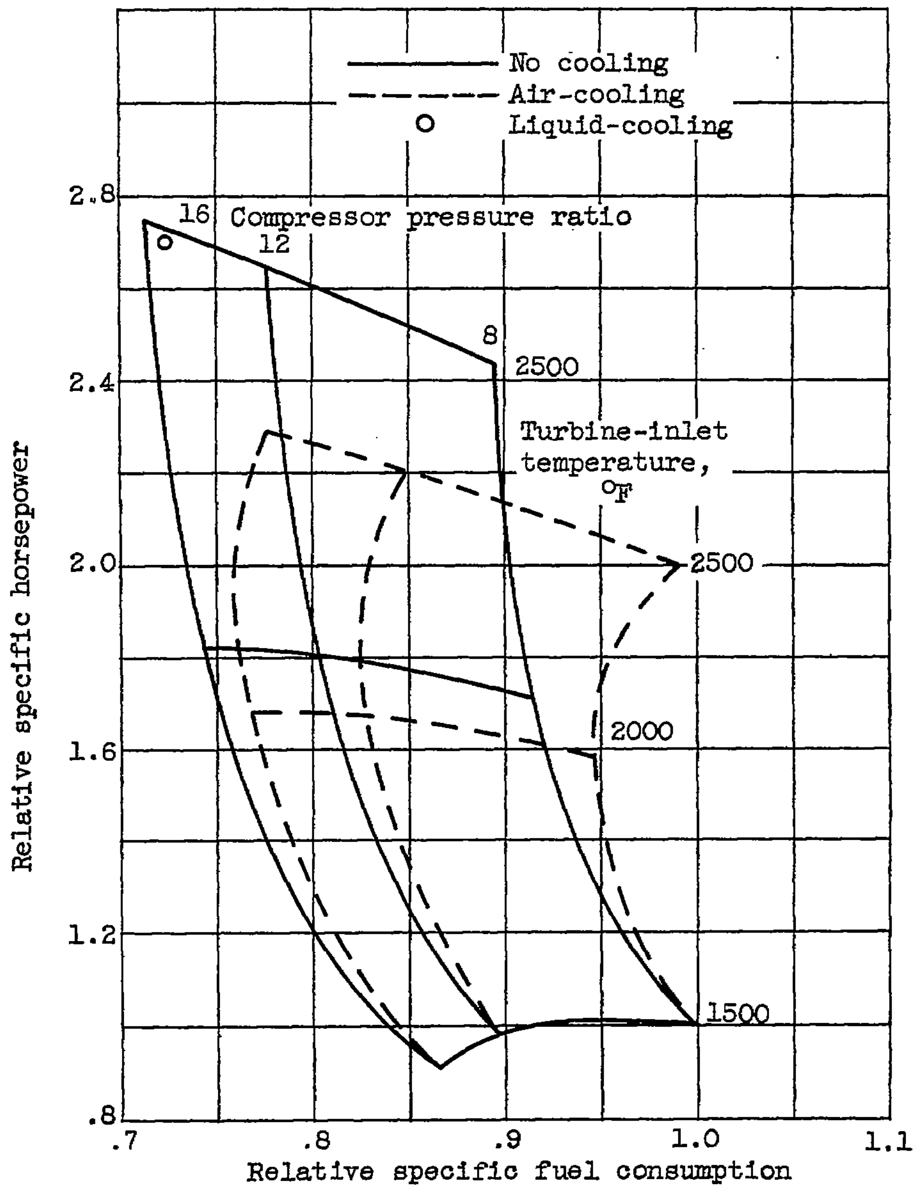

Figure 6. - Effect of air- and liquid-cooling on turboprop-engine performance for ranges of turbine-inlet temperature and compressor pressure ratio at sea-level static conditions.

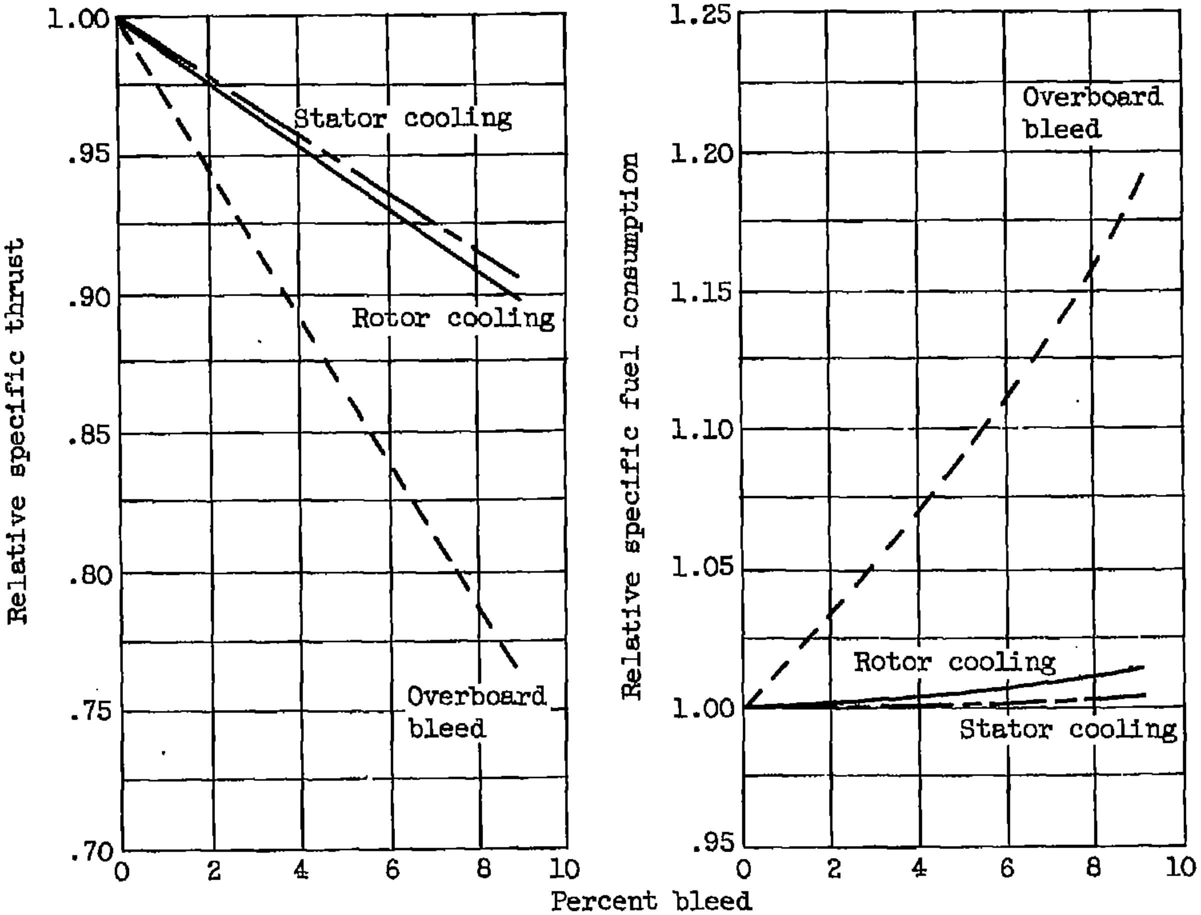

Figure 7. - Variation in performance of nonafterburning turbojet engine with various types of compressor-discharge air bleed. Turbine-inlet temperature, 2000° F; compressor pressure ratio, 6; flight Mach number of 2 in stratosphere.

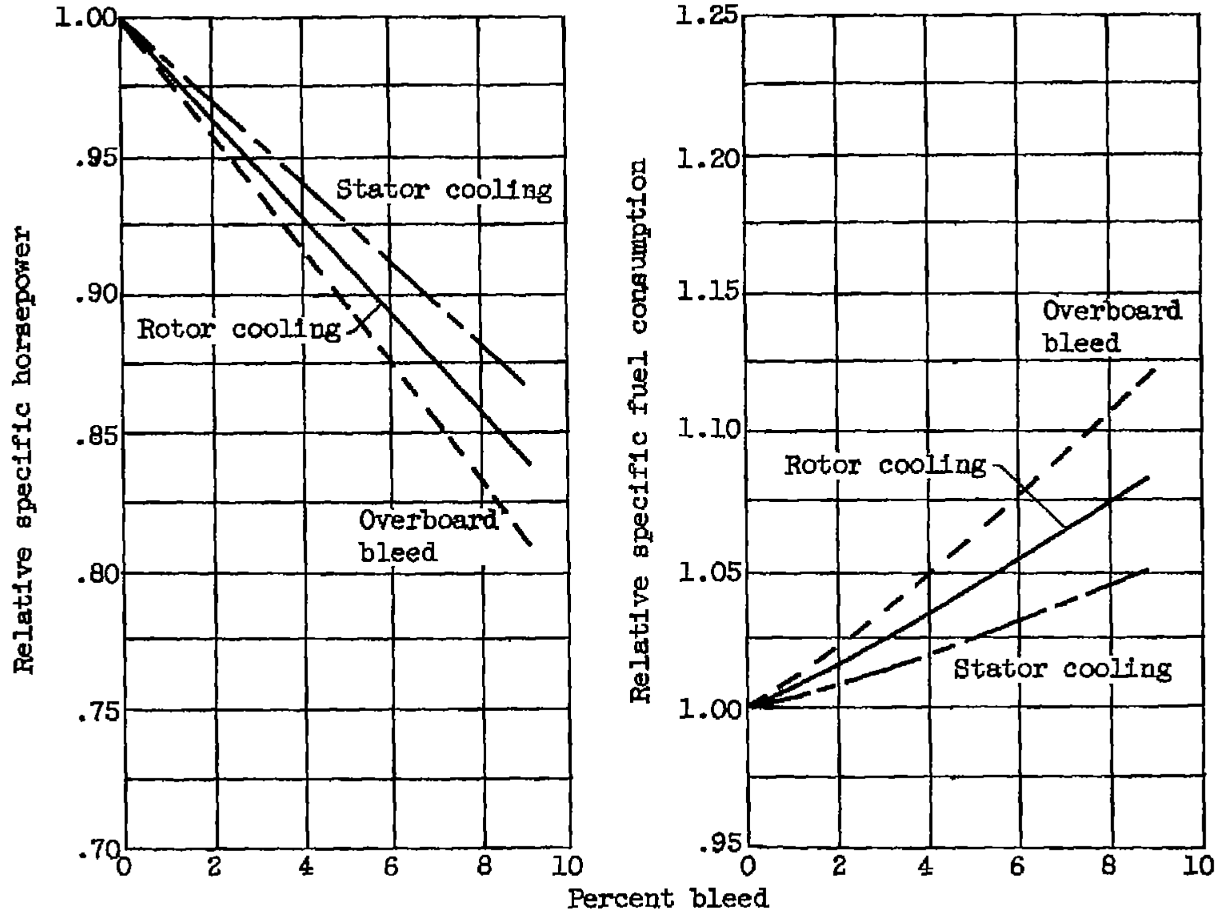

Figure 8. - Variation in turboprop-engine performance with various types of compressor-discharge air bleed. Turbine-inlet temperature, 2000° F; compressor pressure ratio, 12; sea-level static conditions.

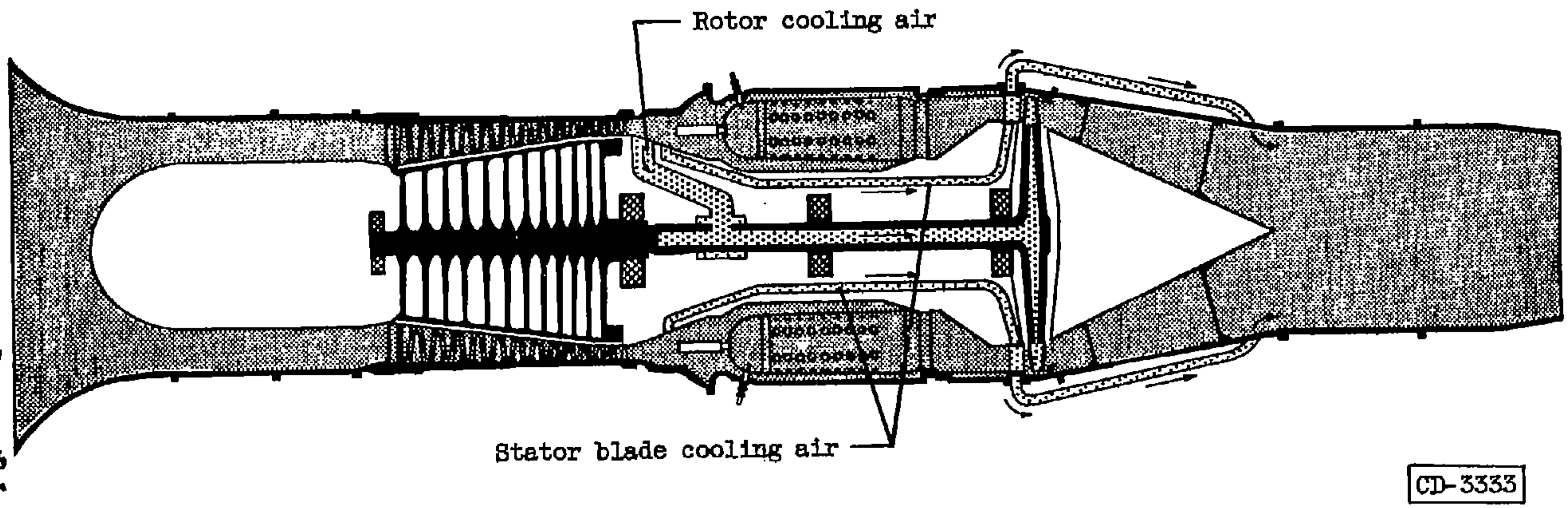

Figure 9. - Cross section of air-cooled turbojet engine.

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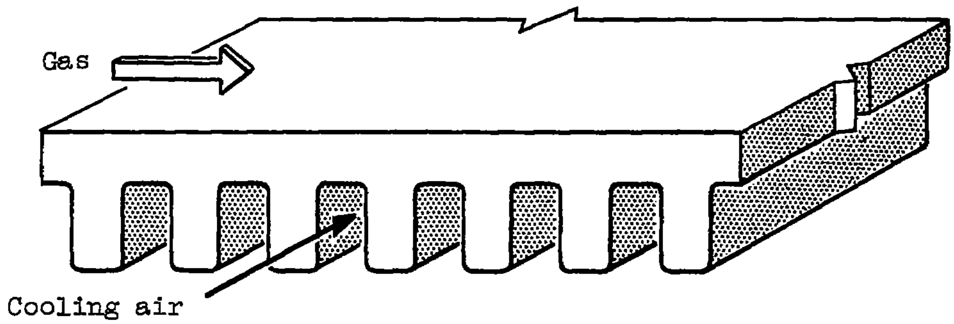
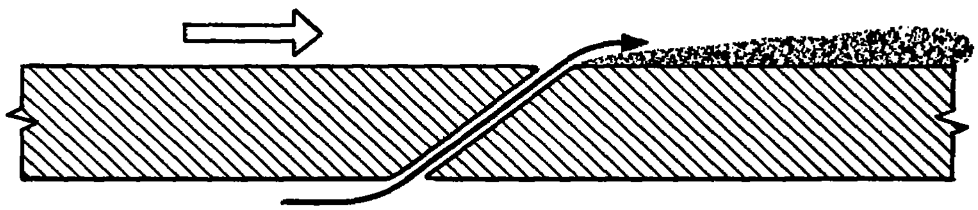
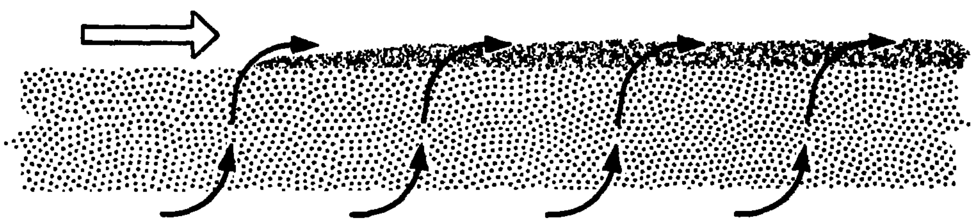


(a) Convection cooling.



(b) Film cooling.



(c) Transpiration cooling.

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Figure 10. - Three methods of air-cooling.

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CA-5 back

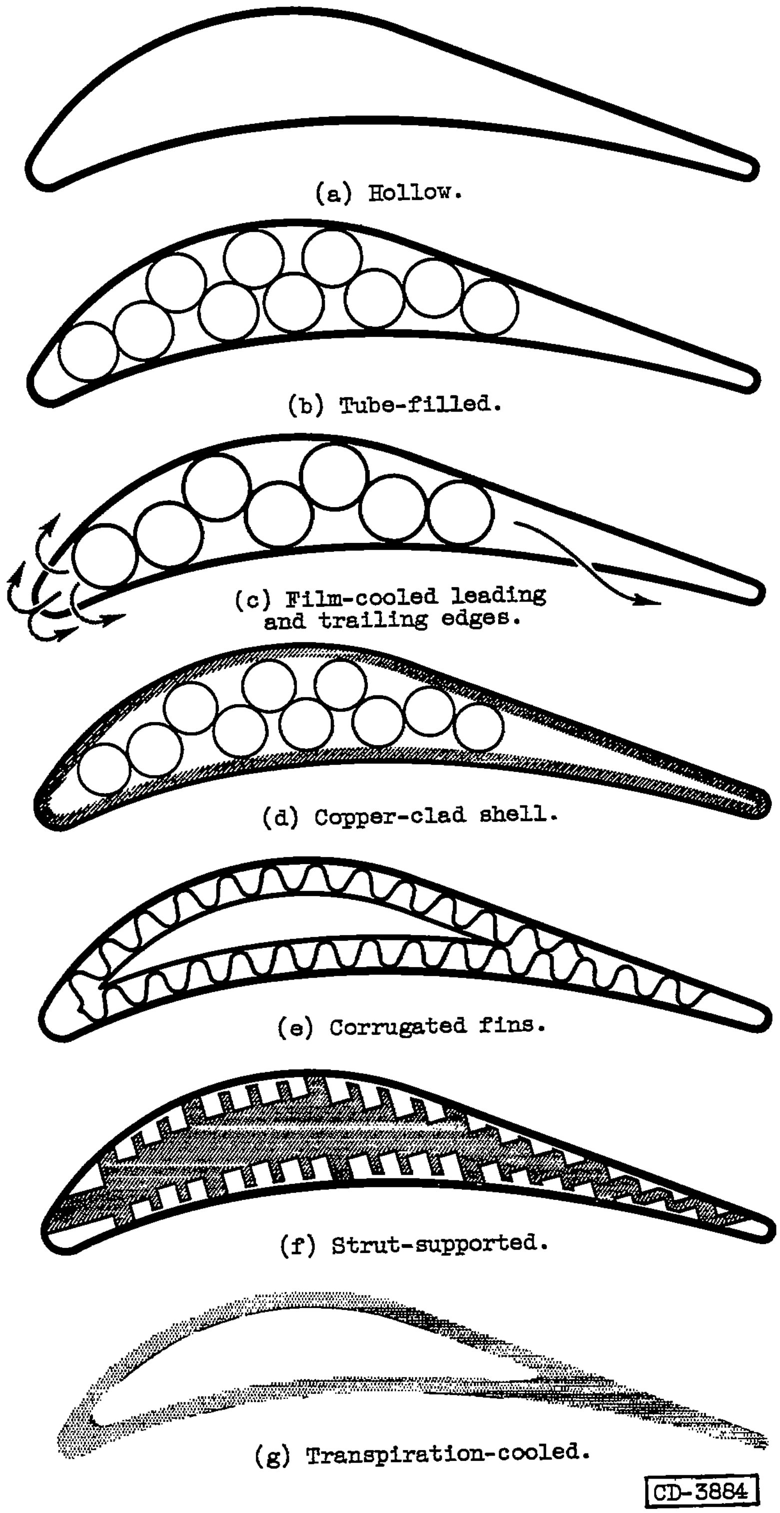

Figure 11. - Air-cooled turbine blade configurations.

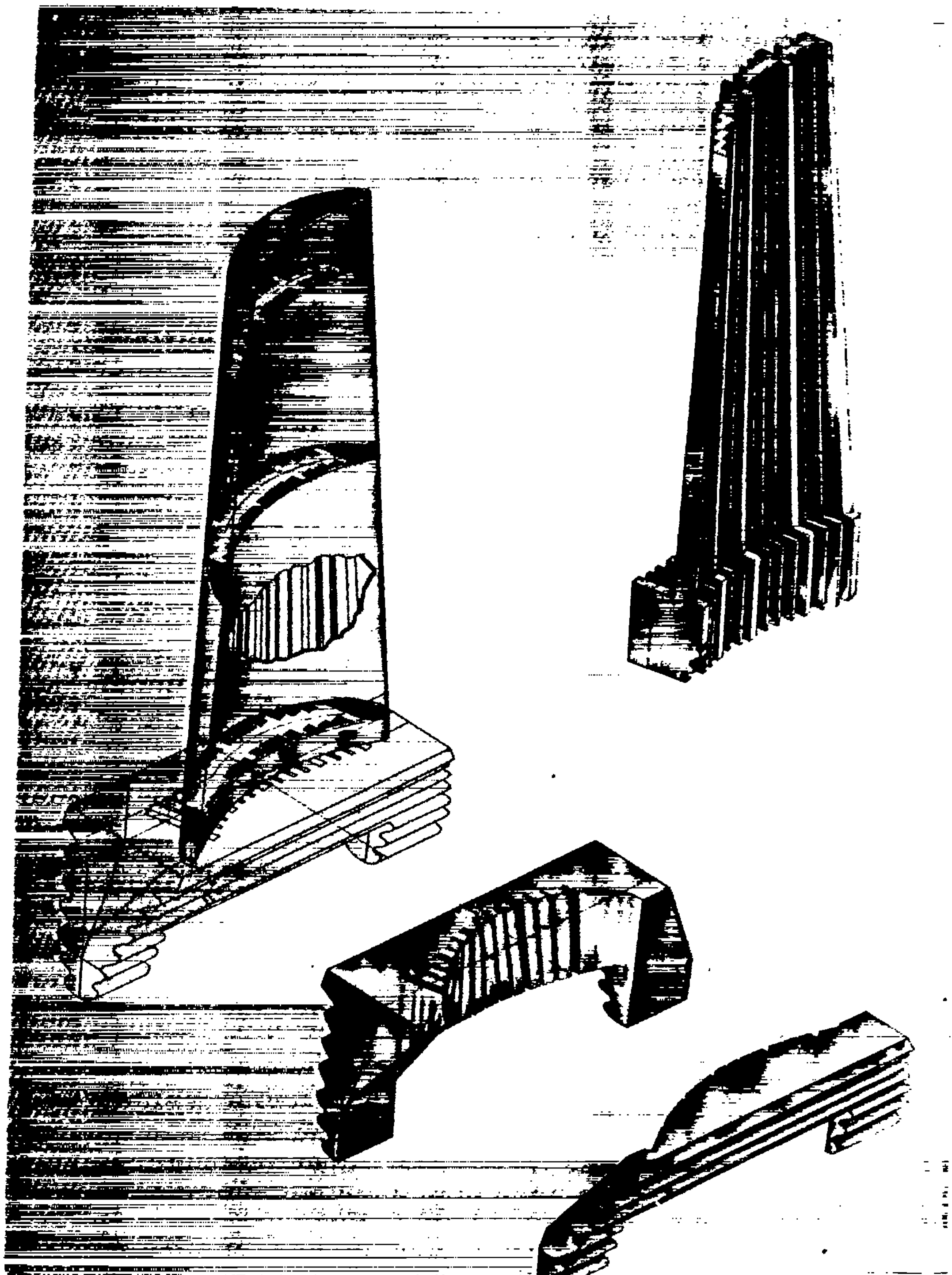

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Figure 12. - Exploded view of air-cooled strut-supported turbine blade.

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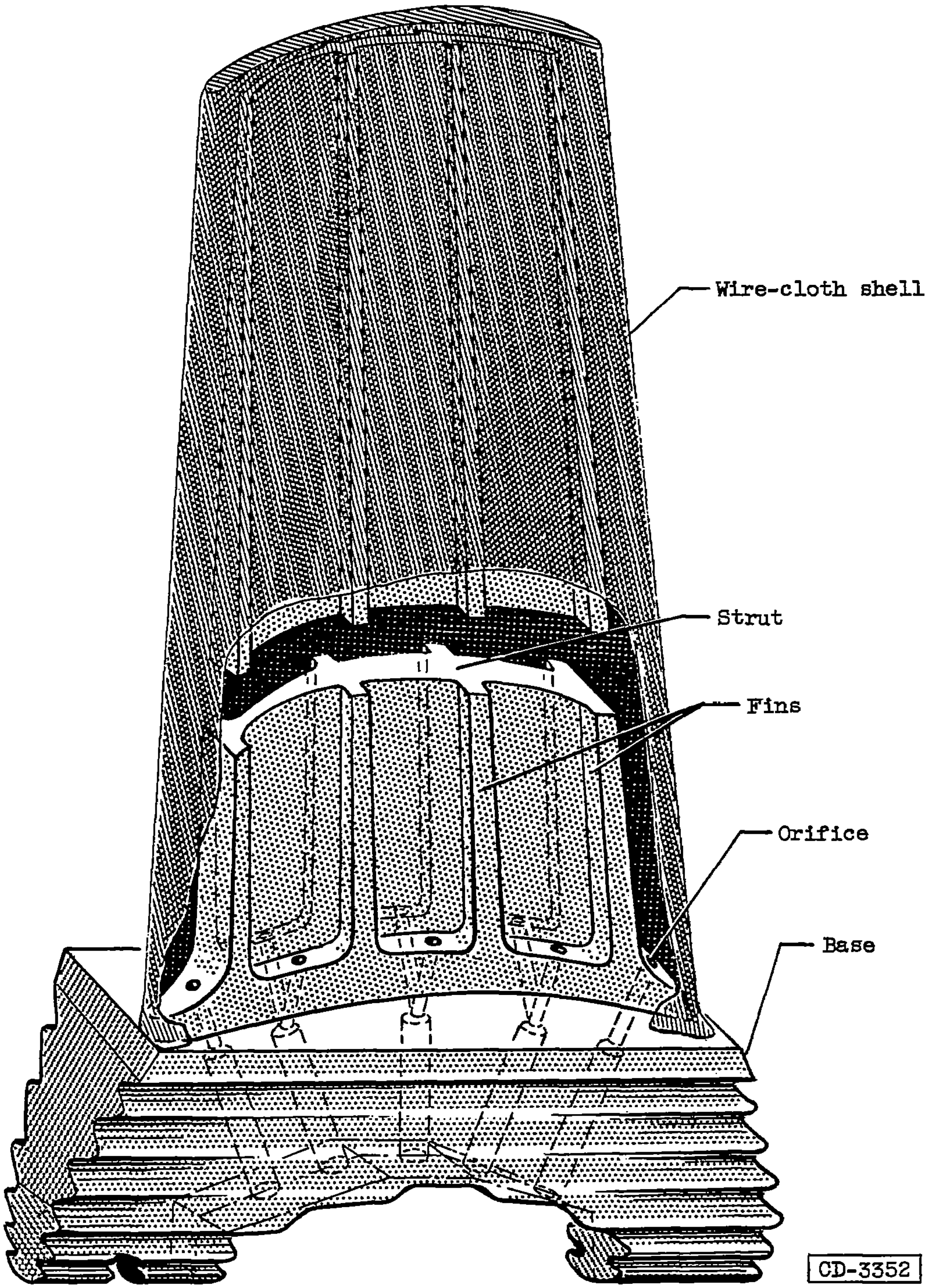

Figure 13. - Sketch of transpiration-cooled blade utilizing orifices in base for metering cooling air.

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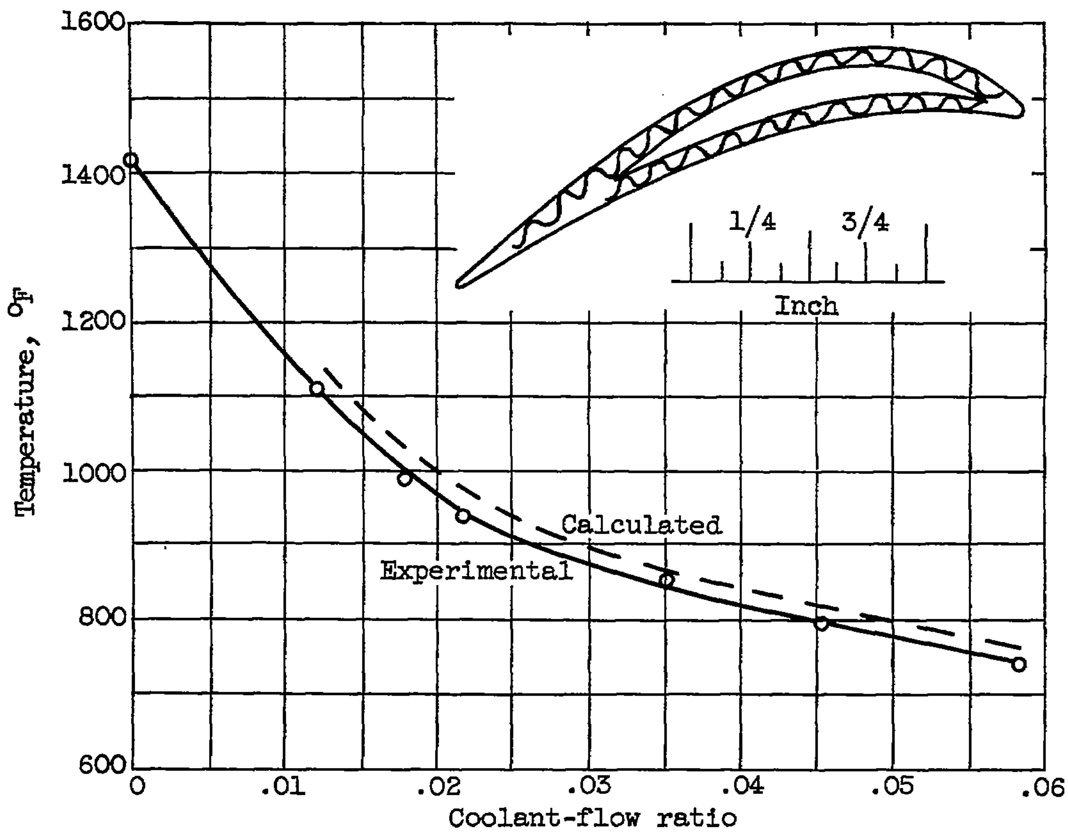

Figure 14. - Comparison of experimental and analytical blade temperatures. Turbine-inlet temperature, 1640° F; corrugated-insert blade.

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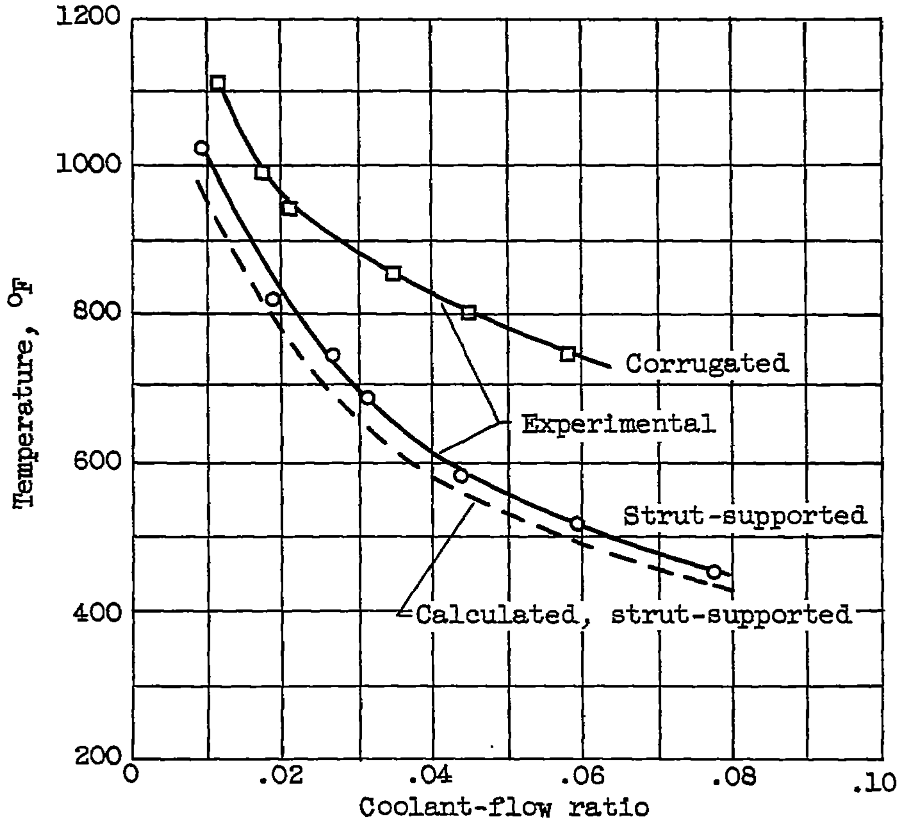
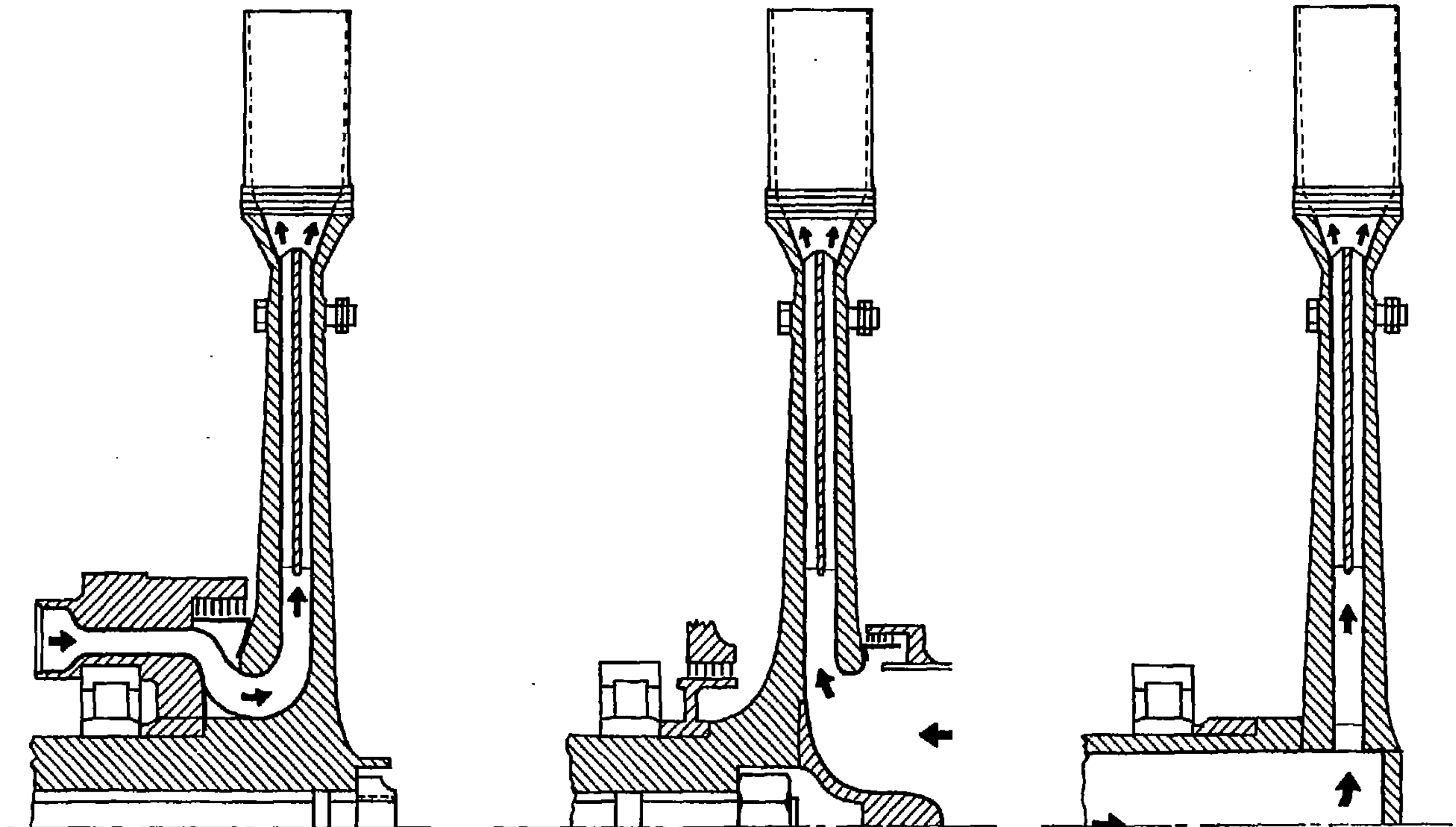


Figure 15. - Temperatures of two convection-cooled turbine blades. Turbine-inlet temperature, 1640° F.



(a) Upstream inlet. (b) Downstream inlet. (c) Hollow-shaft inlet.

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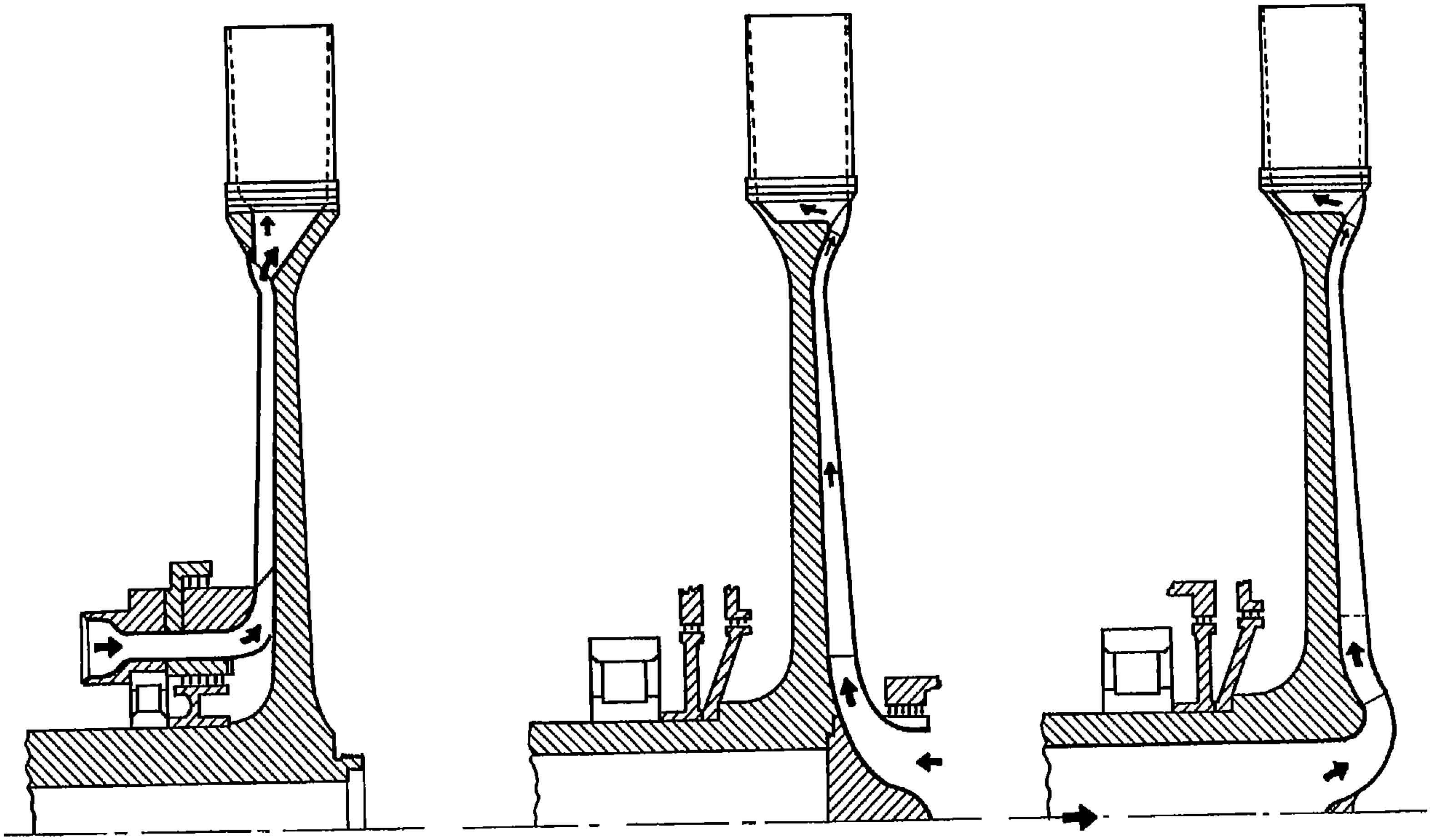
Figure 16. - Split-disk air-cooled turbine configurations.



(a) Upstream inlet. (b) Downstream inlet. (c) Hollow-shaft inlet.

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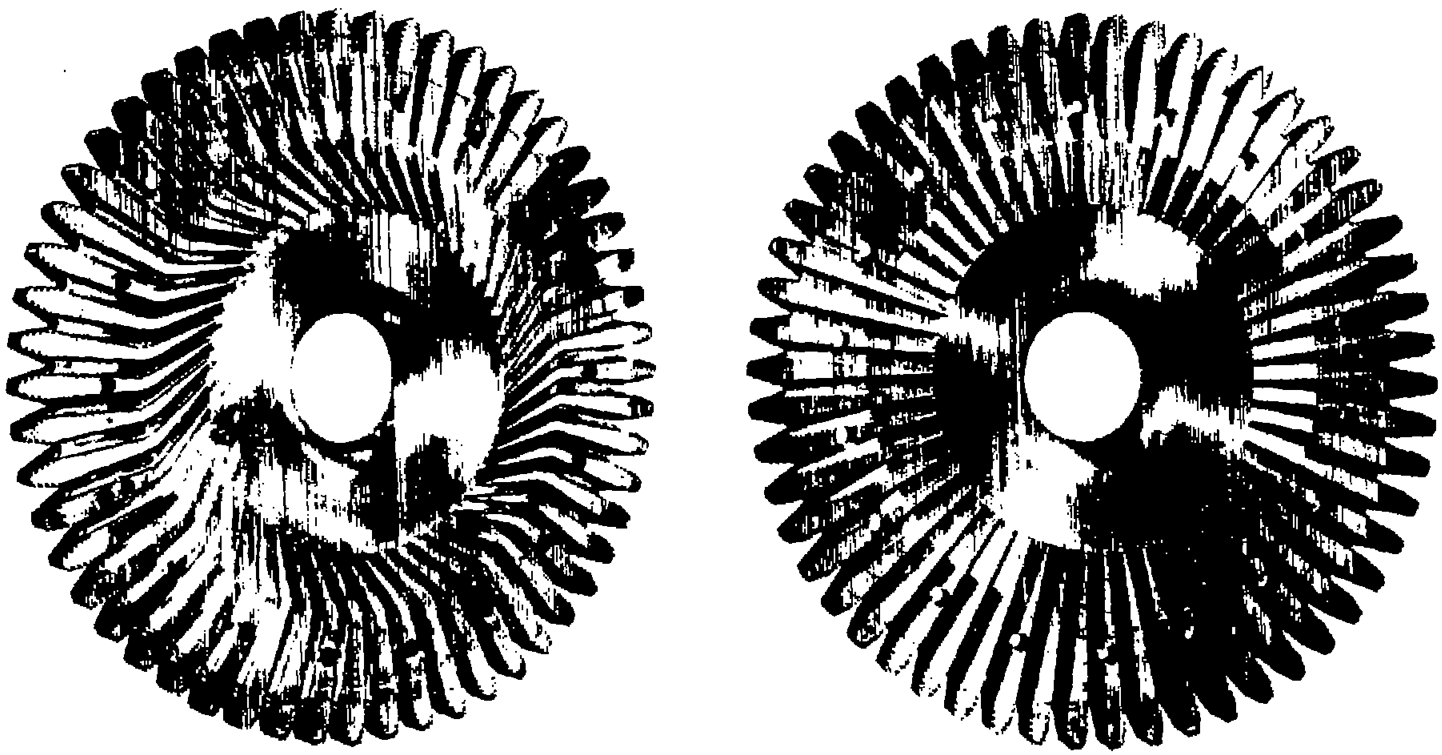
Figure 17. - Shrouded air-cooled turbine disk configurations.



Curved-vane inducer section

Straight vanes

C-33784

Figure 18. - Turbine cooling-air vane configurations.

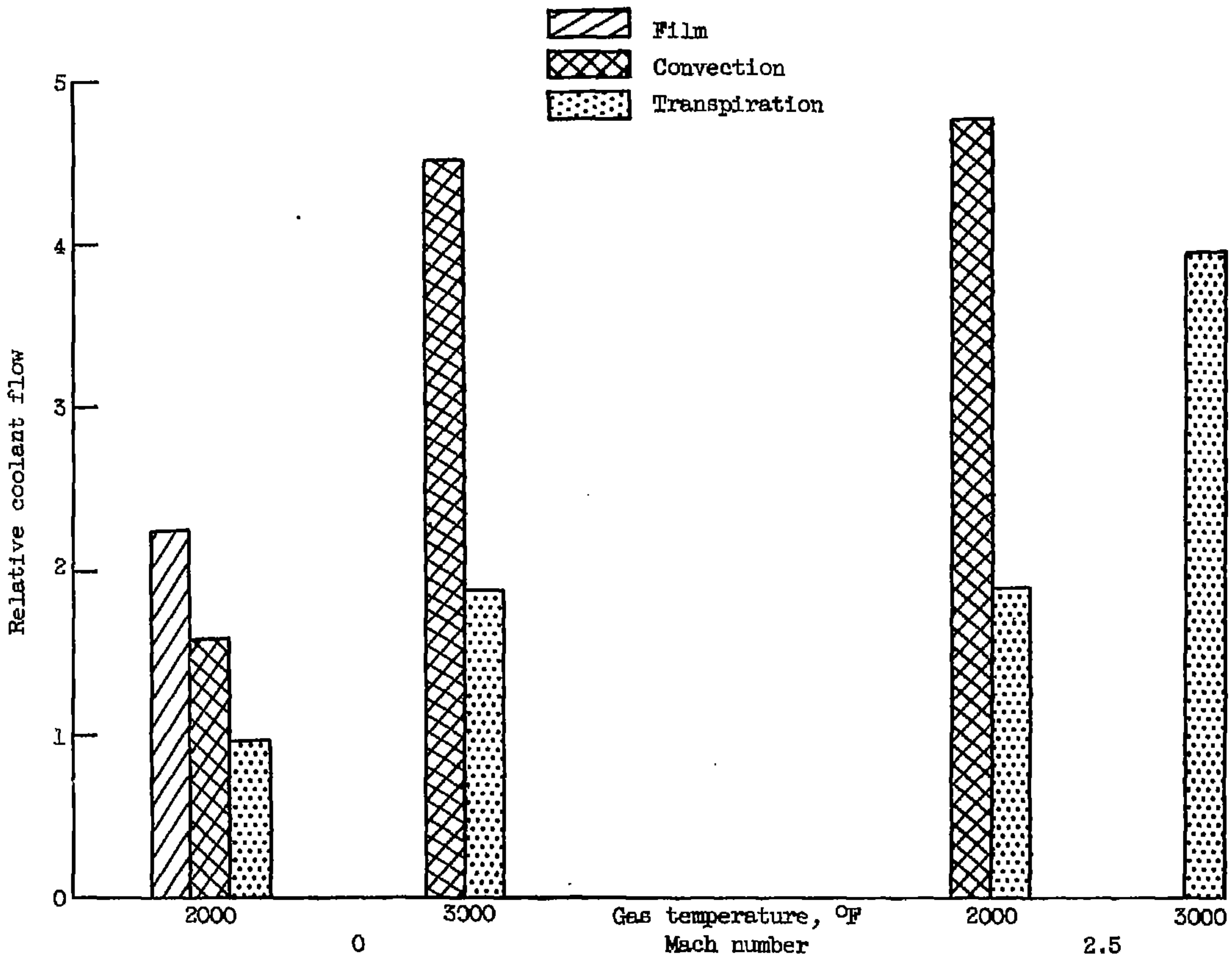


Figure 19. - Comparison of cooling effectiveness of three methods of air-cooling for two turbine-inlet temperatures and flight Mach numbers. Cooling air bled from compressor discharge.